THE EFFECTS OF JET EXHAUST BLAST IMPINGEMENTS ON GRAPHITE-EPOXY COMPOSITES

John Michael Hampey

NAVAL POSTGRADUATE SCHOOL
Monterey, California

THESIS

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BY

John Michael Hampey

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Thesis Advisor D. Salinas

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Item #20 continued:

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The Effects of Jet Exhaust Blast Impingements on Graphite-Epoxy Composites

by

John Michael Hampey
Lieutenant, United States Navy
B.S.M.E., Ohio State University, 1973

Submitted in partial fulfillment of the requirements for the degree of

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The effect of jet exhaust blasts on graphite epoxy composites (Hercules 3501-6/AS4) is examined. The material degradation of the composites is determined by means of the short beam shear test. The jet exhaust tests were designed to test the worst case conditions for an F-18 aircraft operating off an aircraft carrier. Results indicate that the composites show no significant property changes if the temperature is maintained less than 230°C. At temperatures in excess of these, strength degradation occurs. It was also observed that when strength degradation occurs, obvious discoloration and delamination of the composite are evident.

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I. INTRODUCTION

The word "composite" is defined as made up of distinct parts or elements. Composite when used in connection with composite material signifies that two or more materials are combined on a macroscopic scale to form a useful material. The key to distinguising composites from alloys is the macroscopic examination of a material. Different materials can be combined on a microscopic scale, such as alloying, but the resulting material is macroscopically homogeneous. The advantage of composites is that they usually exhibit the best qualities of their constituents and often some qualities that neither constituent possesses. Some properties that can be improved with composite selection include:

- strength
- stiffness
- corrosion resistance
- wear resistance
- attractiveness
- weight
- fatigue life
- temperature-dependent behavior
- thermal insulation
- thermal conductivity
- acoustical insulation

Naturally, not all of the above properties are improved at the same time, nor is there usually any requirement to do so.

Composite materials are usually of three common types:

1. Fibrous composites which consist of fibers in a matrix.
2. Laminated composites which consist of layers of various materials.

3. Particulate composites which are composed of particles in a matrix.

The type of composite of interest in this paper is a fiber reinforced laminated composite. Laminated composites are composed of at least two different materials that are bonded together, the epoxy bonds the graphite (graphite and epoxy in this case). Lamination is used to combine the best aspects of the constituent layers in order to achieve a more useful material. The composite of interest is composed of stiff graphite fibers in a weak ductile epoxy matrix. The properties that can be emphasized by lamination are strength, stiffness, low weight, corrosion resistance, wear resistance, beauty or attractiveness, thermal insulation, acoustical insulation, etc.

The graphite epoxy composite is further classified as a laminated fibrous composite. Laminated fibrous composites are a hybrid class of composites involving both fibrous composites and lamination techniques (also called laminated fiber-reinforced composites). Here, layers of fiber-reinforced material are built up with the fiber directions of each layer typically oriented in different directions to give different strengths and stiffnesses in the various directions, i.e. an anisotropic material.

The composites under study are of the following nominal thicknesses: 1/8 inch, 1/4 inch and 1/2 inch composed of 24, 48, and 96 plies respectively. The above thicknesses were selected as they are typical of composite plates used in

aircraft structures. The ply orientation follows the sequence of 0°, + 45°, 90°, -45° etc. The composite was manufactured by Hitco and is known as Hercules 3501-6/AS4.

The graphite epoxy composite materials are ideal for structural applications where high strength-to-weight and stiffness-to-weight ratios are required. Graphite fibers have approximately half the density of aluminum (.051 lb/in^3 compared to .097 lb/in^3) and nealy three times the tensile strength (250 x 10^3 lb/in^{3}to 90 x 10^3 lb/in^3) and thus their strength-to-weight ratio is six times better than for aluminum [Ref. 13]. This advantage has been recognized and more interest is being focused on the use of composites in aircraft structures. The weight savings of the composites can result in increased aircraft performance, fuel savings, and higher payload then an aircraft built of conventional design. Weight savings of the order of 10 - 30% [Ref. 1] are possible with composites at the present time and future savings could improve considerably, if design is based solely on the use of composites. These increased savings would arise as composites can be designed to achieve the properties desired. Further experience with composites would also greatly enhance the savings as the composites would no longer be designed with excessive factors of safety arising from uncertainties in design. Composites however are not without their problems. One of the problems appears to be the degradation of their physical properties when exposed to high temperature. The cure temperature of the

composite of interest (Hercules 3501-6/AS4) is 177°C. This appears to be the critical temperatures for degradation to occur, as seen in References [2, 3]. Both references show the strength declining above 177°C.

Therefore, the objective of this investigation is to observe the effect of temperature on the strength of the composite and to determine if the critical temperature to initiate degradation would be reached under normal operating conditions. This investigation sought to establish a failure criteria for a particular class of composites based on temperature distribution through the specimen. A series of tests were conducted to investigate a variety of operating conditions. The samples were tested and compared to the baseline reading of the original untested specimens in order to determine the amount of material degradation.

II. NATURE OF THE PROBLEM

Graphite-epoxy fiber reinforced laminate composite materials comprise about 9.9% of the structural weight of the F-18 aircraft (Figure 1) Composite elements include the wing skins, trailing edge flaps, stabilators, vertical tails and rudders, speed brakes and many access doors. The use of these composites has resulted in an appreciable weight savings, plus increased aircraft performance.

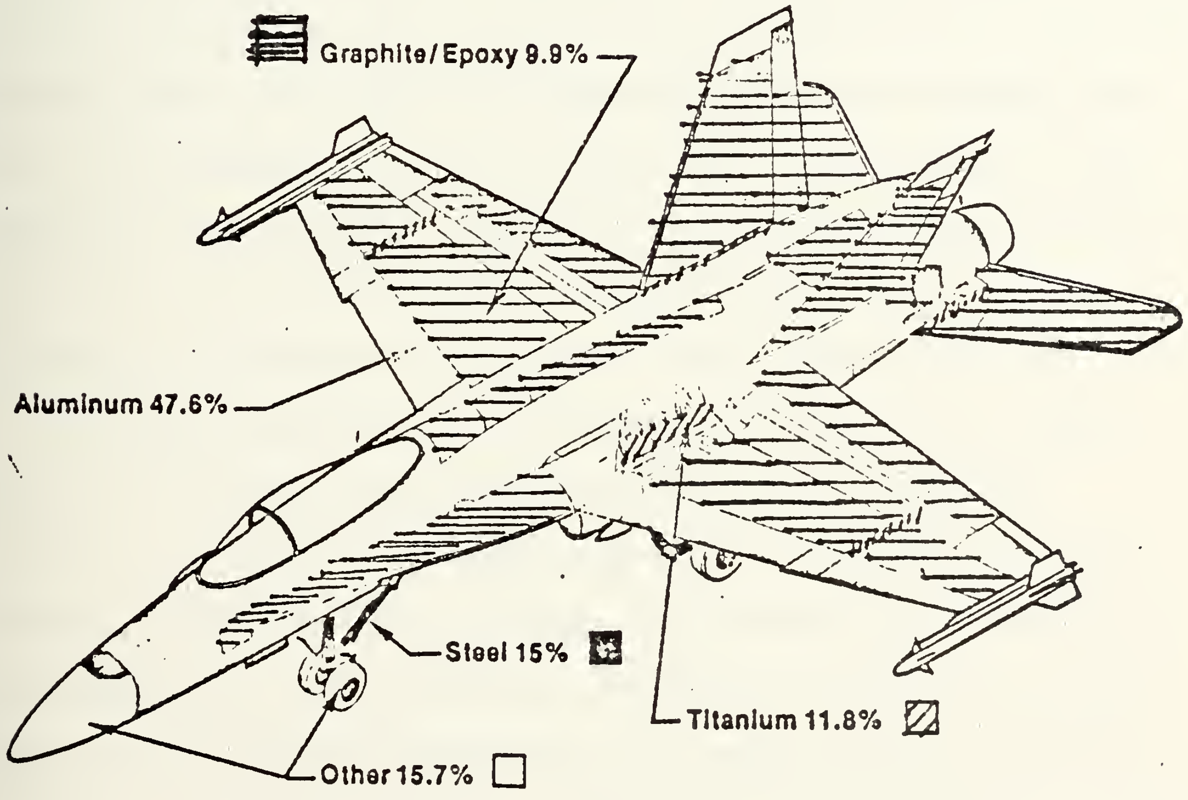


FIGURE 1: MATERIAL COMPOSITION OF F-18 AIRCRAFT

Various studies have shown that the strength of composite materials degrade at temperatures in excess of 177°C (Refs. 2, 3, 6, 7, 9, 11]. Most of the studies were also concerned with moisture content at these temperatures [Refs. 2, 7, 9]. Therefore, these results are not conclusive as to the extent of damage that can be attributed to high temperature alone.

The main objective is to determine whether aircraft composites reach a critical temperature under normal operating conditions. The heat source is the flow of jet exhaust gases from surrounding carrier aircraft. Evaluation of the temperature and velocity profiles (Appendix A) of all aircraft currently operating off a carrier has shown the F-14, to be the critical case. The evaluation consisted of determining heat flux at distances of 10 and 20 feet for all the aircraft and comparing these results. The F-14 was therefore the aircraft chosen for all further computations of heat generation, as it was desired to concentrate on worst case conditions first.

One major obstacle in defining the problem is the determination of normal operating conditions. The operating condition is defined by such parameters as distances between aircraft, power settings, thermal environment, orientation of the composites, and duration of exposure. The NATOPS Flight Manual gives limited information on actual distances between aircraft, and power settings. One source of guidance states that "80% RPM is necessary to set aircraft in motion. Once in motion, idle thrust is sufficient to sustain taxi speeds."

Aircraft taxiing on the deck alongside aircraft waiting in position, present the most hazardous heat condition.

It was originally believed that the worst case condition of heating would occur during launching of the aircraft. This would definitely be the case if not for the Jet Blast Deflector (JBD). The JBD's purpose is to deflect the exhaust gases away from the aircraft waiting for launch. The JBD is a shield approximately 18 feet high, 42 feet wide (made in 3 sections 18 feet by 14 feet) and 9 inches thick. The face of the shield is aluminum, 1-1.25 inches thick. Internally, the JBD is cooled by circulating salt water, provided by the firemain system at 80 psi minimum. The shield deflects the gases over and around the shield, effectively deflecting the direct blast away from the aircraft, on station, waiting for takeoff. References [15, 16, and 17] are concerned with temperature conditons of aircraft waiting behind the JBD. References [15, 16, and 17] show that the critical temperature is not reached under normal launch conditions. The launch condition was therefore, eliminated from consideration.

Distances between aircraft were determined by scale model drawings of the aircraft. Conditions whereby the aircraft could get as close as possible without physically touching, were modelled. The engine power setting was assumed to be able to vary from idle to full power. The duration of exposure was assumed to vary from 2 seconds, minimum exposure, to exposure times necessary to reach a steady state temperature.

Geometric angle of exposure was constant at 0°. The 0° criteria was selected, as sections carrying the most load would always be parallel to the gas flow (0° angle of attack). The geometric angle of exposure is defined to be the angle that the jet blast hits the tested surface: i.e. flow over a horizontal plate is 0° angle of exposure as it is parallel to the flow. The main objective is to establish conditions for failure of the composites. The samples were tested at China Lake in accordance with Appendix B. The thermal environment was created by jet engine blast. The resulting samples were sent to Naval Postgraduate School for short beam shear tests as per ASTM D-2344 (Appendix C).

The original samples were all flat plates, 6 inches by 6 inches, manufactured by Hitco Corporation. The samples had nominal thicknesses of 1/8 inch, 1/4 inch and 1/2 inch prior to testing. All samples were painted with the same paint used on the F-18 aircraft.

III. EXPERIMENTAL PROCEDURES

A. ESTABLISHMENT OF A FAILURE CRITERIA

1. Calculations

The first problem to be resolved was the temperature at which the composite material should be considered as failed. This was accomplished by testing samples in a controlled heating situation by use of an oven where the heat flux could be closely controlled. The heat flux to the samples was based on calculations for five possible situations (Table I). Reference [4] was used for obtaining the necessary equations for calculation of thermal conductivity for flow over a flat plate:

$$\overline{Nu}_L = \frac{\bar{h}L}{k} = Pr^{1/3}(0.037\ Re_L^{0.8} - 850) \qquad (1)$$

$$\frac{q}{A} = \bar{h}\ (T_A - T_\infty) \qquad (2)$$

The values obtained from these equations were compared to calculations obtained using the equations of Reference [5].

$$St = \frac{Nu_x}{Re_x Pr} = \frac{0.0297 Re_x^{-1/5}}{1 + 1.48 R_e^{-1/16} Pr^{-1/6}(Pr - 1)} \qquad (3)$$

Both solutions were in good agreement but Holman's [Ref. 4], equations consistently gave slightly higher values (eq. 1.35 x 10^2 as compared to 1.11 x 10^2 $\frac{W}{cm^2}$). The higher values obtained by Holman's equations were used. Table I lists the engine setting, distance and heat flux which the controlled situation attempted to duplicate.

TABLE I

HEAT FLUX DETERMINED BY CALCULATION

ENGINE SETTING	DISTANCE FROM EXHAUST	q/A ($\frac{W}{cm}$)
IDLE	5 FEET	1.35 (1.189 $\frac{Btu}{ft.}$ sec.)
80% MILITARY POWER	5 FEET	7.64 (6.73 $\frac{Btu}{ft.}$ sec.)
90% MILITARY POWER	10 FEET	12.9 (11.36 $\frac{Btu}{ft.}$ sec.) 14.2 (12.5 $\frac{Btu}{ft.}$ sec.)*
80% MILITARY POWER	10 FEET	6.44 (5.67 $\frac{Btu}{ft.}$ sec.)
80% MILITARY POWER	20 FEET	4.54 (3.99 $\frac{Btu}{ft.}$ sec.)

* Value of 14.2 obtained using Holman's relations for Reynolds number above 10 (Re= 1.17 x 10^7 for this case) Relationships are:

$$\bar{C}_f = \frac{0.455}{(\log Re_1)^{2.584}} - \frac{1700}{Re_1} \qquad (4)$$

$$St_x \, Pr^{2/3} = \frac{\bar{C}_f}{2} \qquad (5)$$

Where $St_x = \frac{\quad}{C_p U}$

SAMPLE CALCULATION FOR TABLE I

The engine setting was 80% military power, with the composite located 10 ft. from the engine exhaust, and parallel to the gas flow.

GIVEN:

Temperature of exhaust gases (375 F) 190.5 C

Velocity of exhaust gases (527 MPH) $235.59 \frac{M}{sec} = U_o$

$$T_f = \frac{190.5 + 21.1}{2} = 105 \text{ C} = 378 \text{ K}$$

Values for air used were taken at 400 K Table A-5 [Ref. 4]

$\rho = .8826 \frac{kg}{m}$ $\qquad C_p = 1.014 \frac{KJ}{kg\ C}$

$\mu = 2.286 \times 10^{-s} \frac{kg}{m \cdot s}$ $\qquad K = 0.03365 \frac{W}{m\ c}$

$Pr = .689$

CALCULATED:

$$Re_L = \rho \frac{U\ L}{\mu} = \frac{(.8826)\ (235.59)\ (1)}{2.286 \times 10^{-5}} = 9.09 \times 10^6$$

$$\overline{NU}_L = \frac{L}{k} = Pr^{1/3}(0.037\ Re_L^{\ 0.8} - 850) = 1.13 \times 10^4$$

$$\bar{h} = Nu_L \frac{k}{1} = 1.13 \times 10^4 \left(\frac{.03365}{1}\right) = 3.8 \times 10^2 \frac{W}{m^2\ C}$$

$$\frac{q}{A} = \bar{h}\ (T_a - T_w) = 3.8 \times 10^2\ (190.5 - 21.1) = 6.44 \times 10^4 \frac{W}{m^2}$$

$$\frac{q}{A} = 6.44 \frac{W}{cm^2}$$

2. Laboratory Testing of Samples

Prior to any testing the composites had thermocouples mounted through the sample thickness in accordance with Table II. The thermocouples were installed by drilling holes from the back face to the necessary depths to locate them as desired from the front face.

TABLE II

THERMOCOUPLE LOCATIONS

SPECIMEN THICKNESS	LOCATION T_2(IN)	LOCATION T_2(IN)	LOCATION T_3(IN)	LOCATION T_4(IN)
1/8 INCH	1/16	BACK FACE	N.I.	N.I
1/4 INCH	1/16	1/8	BACK FACE	N.I
1/2 INCH	1/16	1/8	1/4	BACK FACE

Note: Locations are all measured from the exposed (front) face of the composite to the back face in inches N.I. means not installed.

The oven tests were conducted at China Lake under the supervision of John S. Fontenot. Only the 1/4 inch samples were tested due to failure of the oven, and the necessity to proceed to the jet blast test.

The heat flux of the oven was tested for steady state operation via installed thermocouples prior to insertion of the samples in the oven. The oven had an electrical heating element in the roof. The samples were exposed directly under the heating element. The sides and bottom of the samples were insulated with fibrafax so the heat transfer would take place from the top (exposed face) surface inward to the insulated face.

The oven was only capable of generating $4.0 \frac{W}{cm^2}$ of heat flux. It was therefore not possible to duplicate the upper readings of Table I. Three samples were subjected to 3 different heat fluxes. The criteria for removing two of the samples was when thermocouple T_2 (Table II) reached 200°C. The third sample was to remain in the oven until a steady state temperature was reached. However, after the 5 minutes, temperature T_1 reached 300°C and the sample began to smolder. The sample was removed at this time. The samples, heated to 200°C exceeded the critical temperature, thought to be 177°C. At the Naval Postgraduate School the samples were cut as per Table III, (Figure 2) and then tested, in accordance with ASTM D 2344 (Appendix C: Apparant Horizontal Shear Strength of Reinforced Plastics by Short Beam Method).

TABLE III

TEST SPECIMEN SIZES FOR SHORT BEAM SHEAR TEST

SPECIMEN THICKNESS	WIDTH	LENGTH	TEST SPAN
1/8 INCH	1/4 INCH	8/10 INCH	1/2 INCH
1/4 INCH	1/4 INCH	3/2 INCH	1 INCH
1/2 INCH	1/4 INCH	3 INCH	21/10 INCH

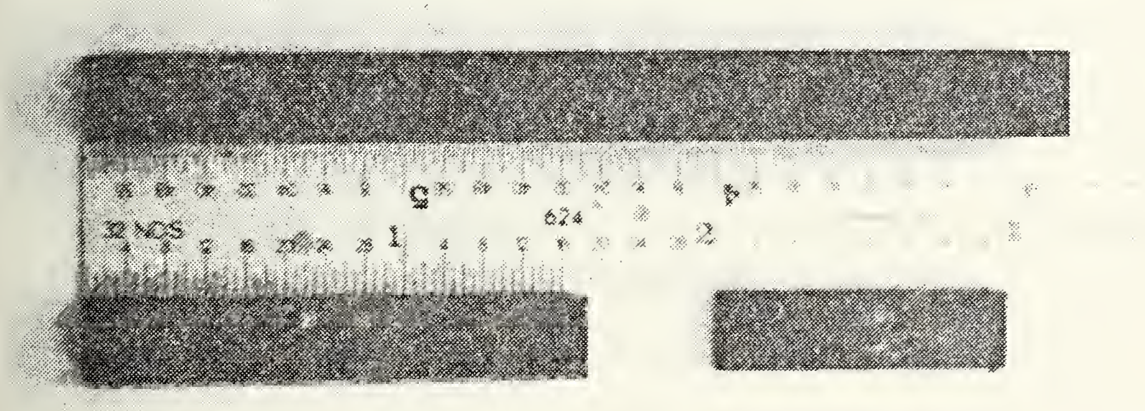


FIGURE 2: COMPARATIVE LENGTHS OF 1/2, 1/4, AND 1/8 INCH SAMPLES RESPECTIVELY FROM TOP, BOTTOM LEFT, AND RIGHT

The procedure followed in testing the samples by the three point short beam shear are outlined on page in section III D.

B. DETERMINING JET BLAST TESTS

Numerous specimen parameters such as thickness and porosity as well as engine exhaust conditions will affect the response of the composite. Thickness and type of paint are the main specimen-controlling parameters. The effect of the paint is due to the different emissivities of the various colors. The tests were designed to investigate a wide range of parameter values. All specimen thicknesses were exposed to the same thermal conditions. The paint selected was grey due to being the worst case situation as far as paint type. The controlling experimental parameters are engine type, engine power setting, distance between the engine and specimen, time duration of exposure and specimen angle in the exhaust flow. The TF-30 engine used was mounted on an F-111 aircraft. This is the same engine installed on the F-14 but no F-14's were available for testing

Time constraints along with the limited funds determined that the distance bewteen jet exhaust and specimen could not be varied at this time. The distance was therefore set at 10 feet, which was determined to be the worst case condition. The exposure times were varied from 2 seconds to a time when the composite reached a steady state temperature. The engine power setting was varied between idle and 90% military power. The angle of attack was fixed at zero degrees (the composite panel was parallel to the exhaust flow). This variable was

also fixed due to time and cost constraints. The selected variables are the worst case conditions believed obtainable in normal operational conditions. With the exception of the angle of attack, the worst angle of attack condition would be the 90 degree case. This condition was not taken for two reasons; 1) the composite panels at this angle are not major load carrying members, and 2) they are at distances greater than the 10 feet worst case condition.

It is appropriate to again point out that the tests that follow is not intended to be inclusive, but is designed to simulate the most severe real world conditions. A follow on study is planned for the China Lake group to go aboard a carrier to measure flow rates and temperatures at various points on the aircraft during aircraft operations. A study will also be conducted to determine actual operating distances between aircraft. These studies, when completed, will enable a more refined and accurate test matrix to be developed.

C. TEST PROCEDURES FOR EXPOSING COMPOSITES TO JET ENGINE BLAST

Appendix B, contains the test plan for jet engine blasts exposures developed by China Lake. A few modifications were made to this procedure and the actual procedure is as follows:

1. Photograph and weight each test specimen(s). (Code 3383).

2. Connect wing box thermocouple leads to specimen(s) thermocouples.

3. Mount test specimen(s) in wing box. (Figure 5)

4. Start TF-30 engine and warmup at IDLE power.

5. Accelerate engine to desired power setting.

6. For a given test condition extending over several days, make final engine power adjustments to maintain a fixed EGT (engine exhaust temperature).

7. Move wingbox into place (Figures 9 and 10)

8. Record test start time.

9. Record all engine parameters.

10. Monitor and record specimen thermocouple readings. When they reach predetermined temperature or predetermined time has elapsed, return engine power to IDLE.

11. Remove wingbox from jet blast (Figures 7 and 8)

12. Continue to record specimen temperatures until they reach ambient temperature.

NOTE: If composite specimen is burning at end of test, extinguish with water, avoid breating of smoke from such specimens.

13. Shut down engine.

14. Shut off recorders.

15. Photograph test specimen(s). (Figures 11 and 12)

16. Allow specimen to cool.

17. Unbolt and remove test specimens, taking care not to further damage heat exposed face.

18. Place specimen in zip-lock bag with card identifying the specimen and the conditions it was tested at.

19.) Weigh specimen (Code 3383).

20.) Store specimen for Project Engineer.

NOTE: All on-site personnel handling test specimens after exposure to jet blast shall wear protective clothing per O.P. 3184-8.b, dtd 25 June 79.

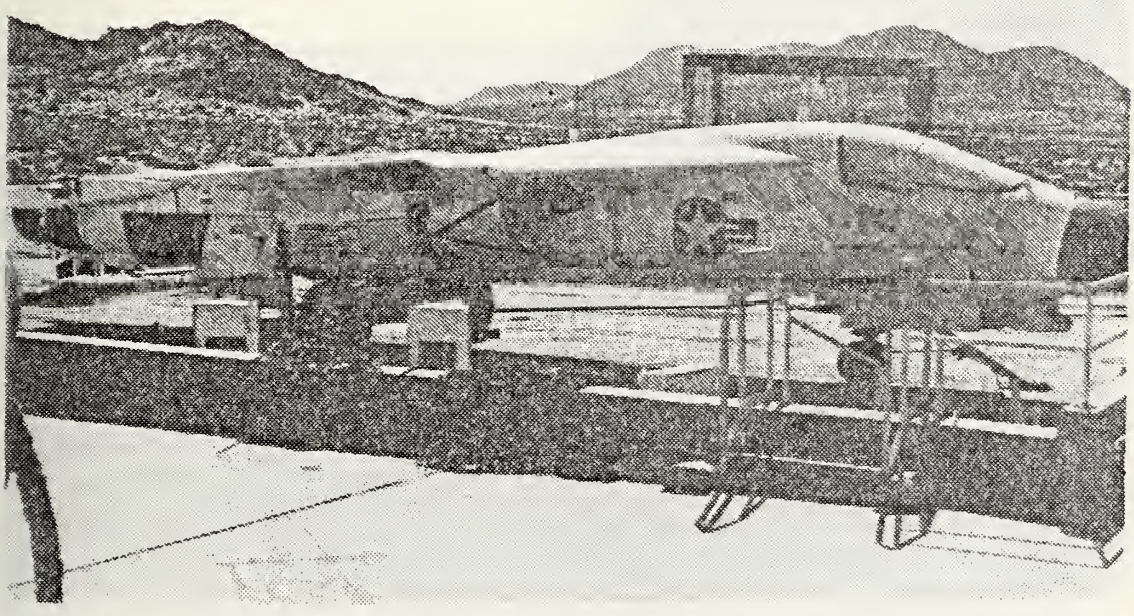


FIGURE 3: F-III AIRCRAFT USED FOR JET ENGINE BLAST TESTING

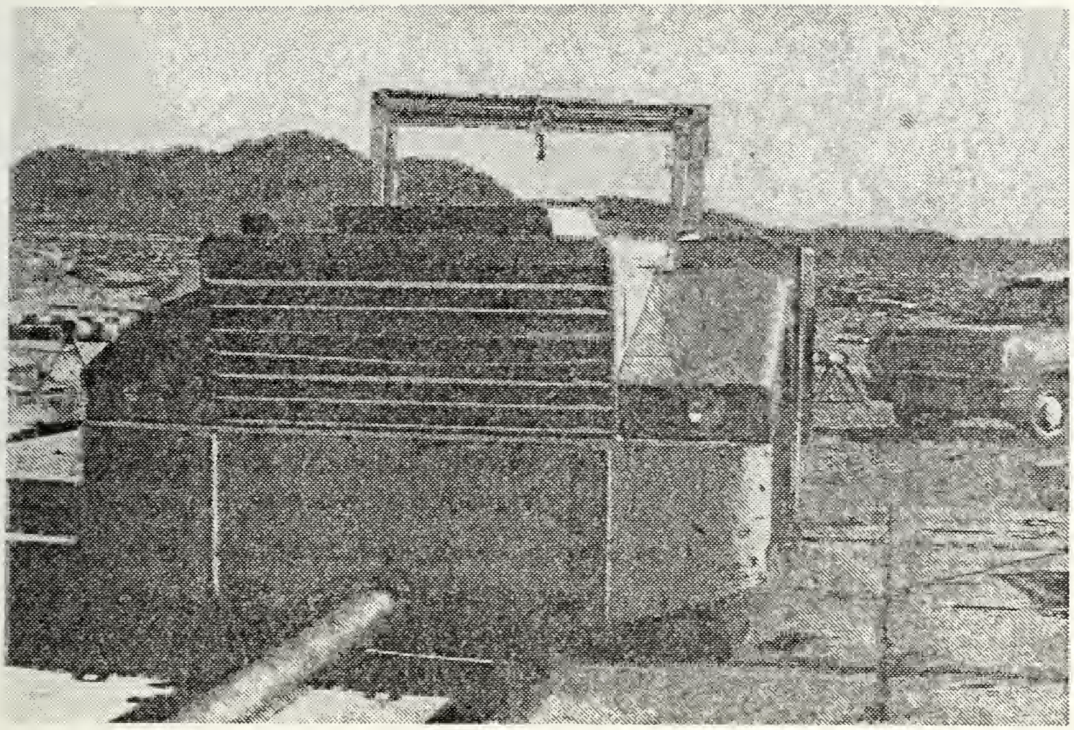


FIGURE 4: VEHICLE USED TO POSITION WINGBOX FOR TESTS

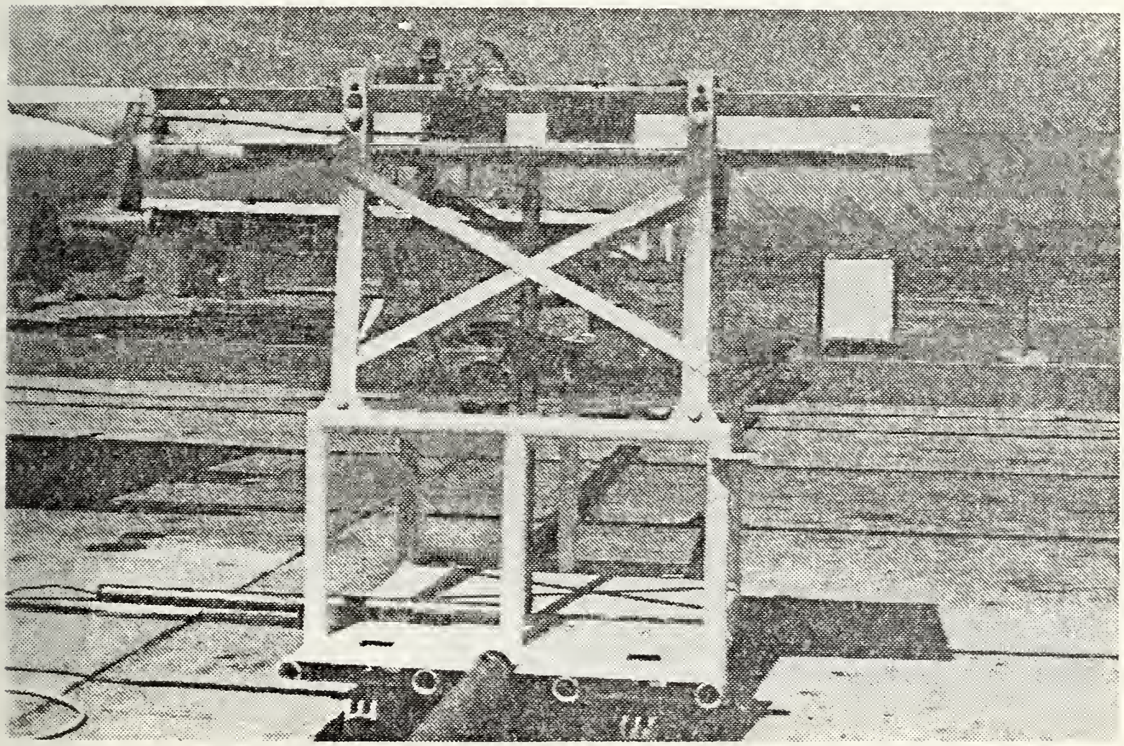


FIGURE 5: WINGBOX USED TO HOLD SPECIMENS DURING TESTING

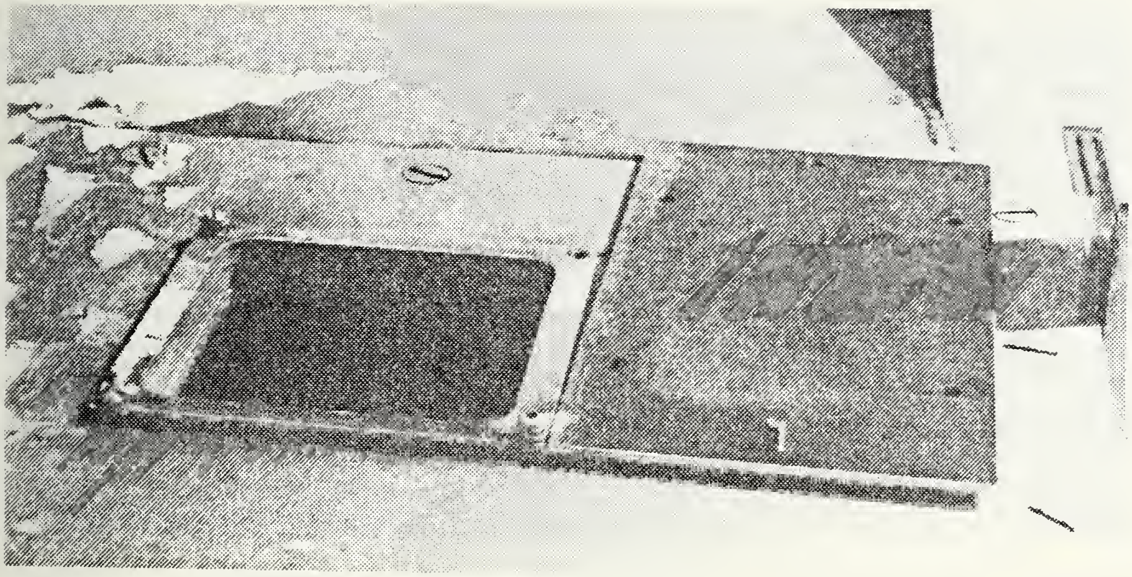


FIGURE 6: CUTOUT FOR MOUNTING SPECIMENS

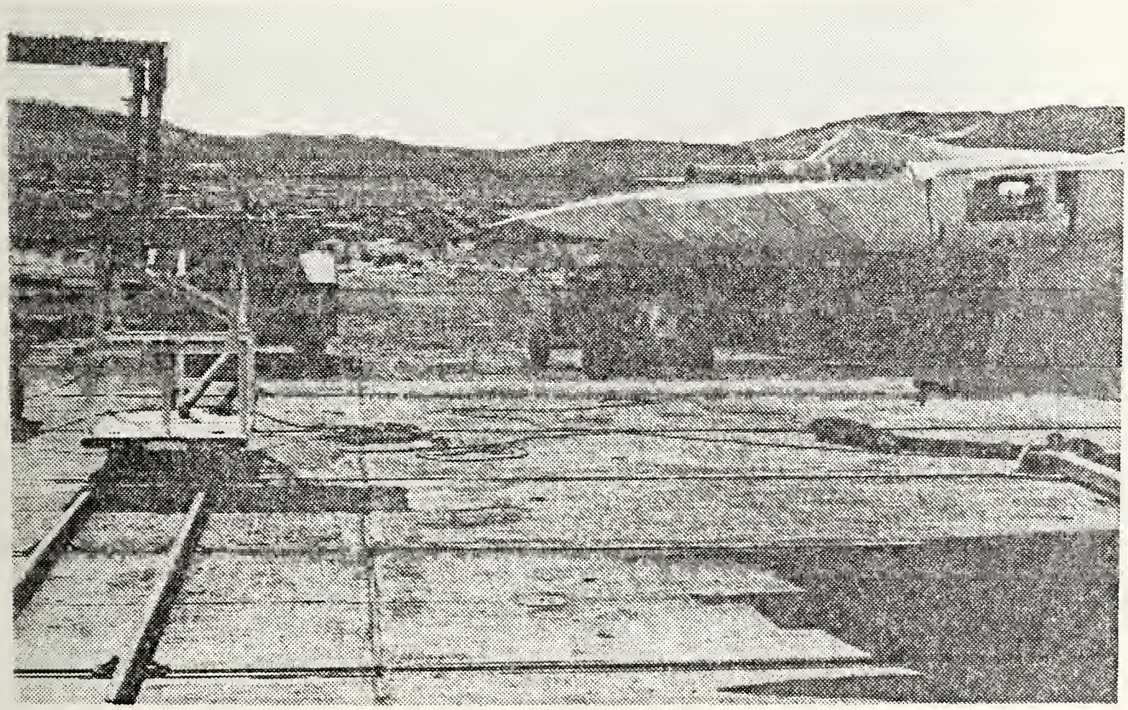


FIGURE 7: POSITION OF WINGBOX BEFORE AND AT CONCLUSION OF EXPOSURE TO JET BLAST

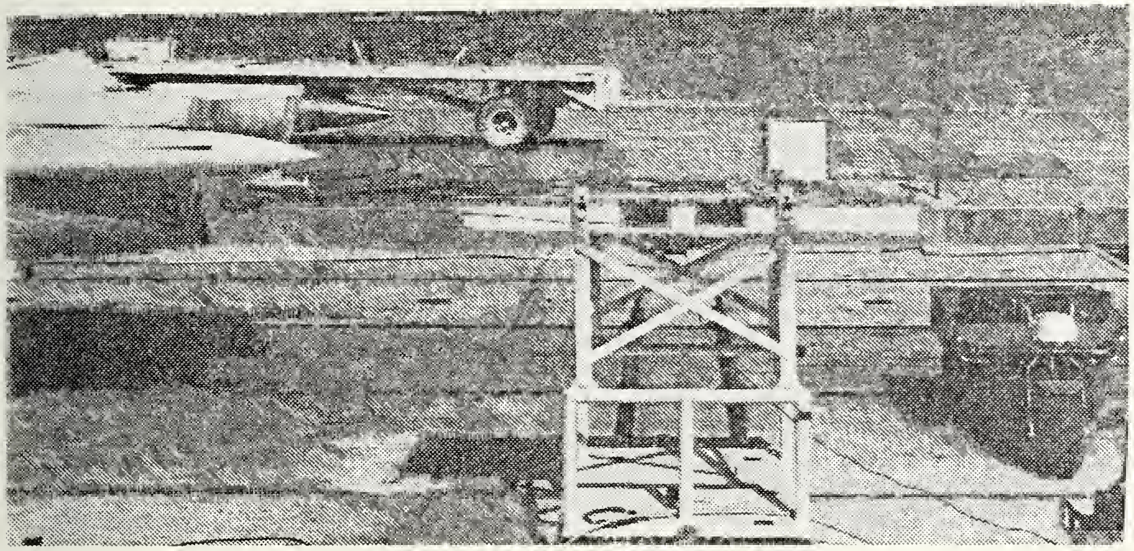


FIGURE 8: ANOTHER VIEW OF POSITION OF WINGBOX PRIOR TO AND AT CONCLUSION OF TESTING

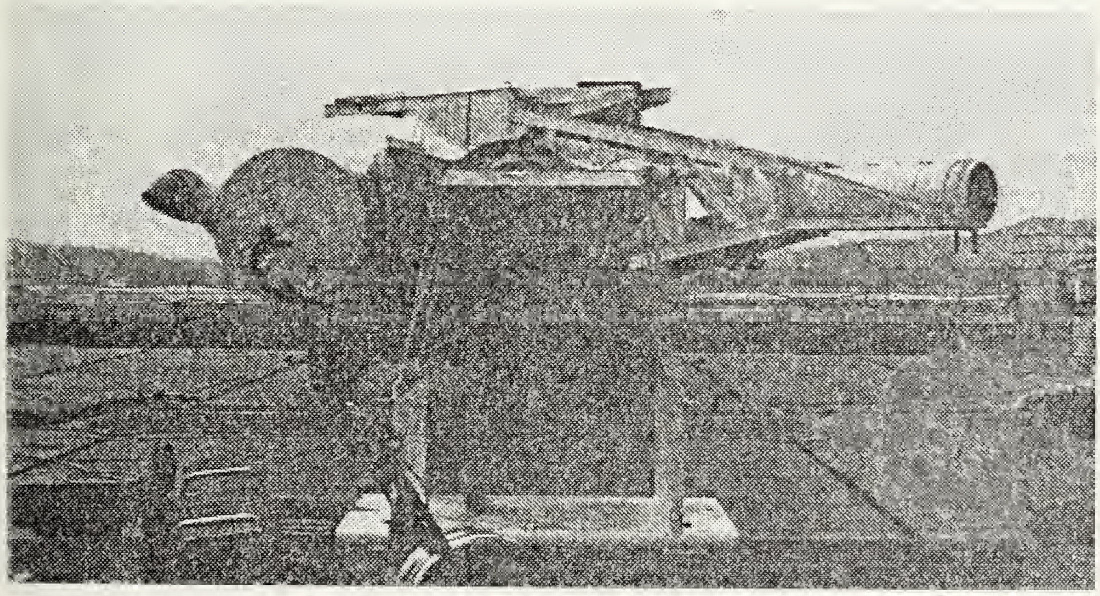


FIGURE 9: LOCATION OF WINGBOX DURING EXPOSURE TO JET BLAST (HEAD VIEW)

FIGURE 10: LOCATION OF WINGBOX DURING EXPOSURE TO JET BLAST (VIEW 45° ASPECT)

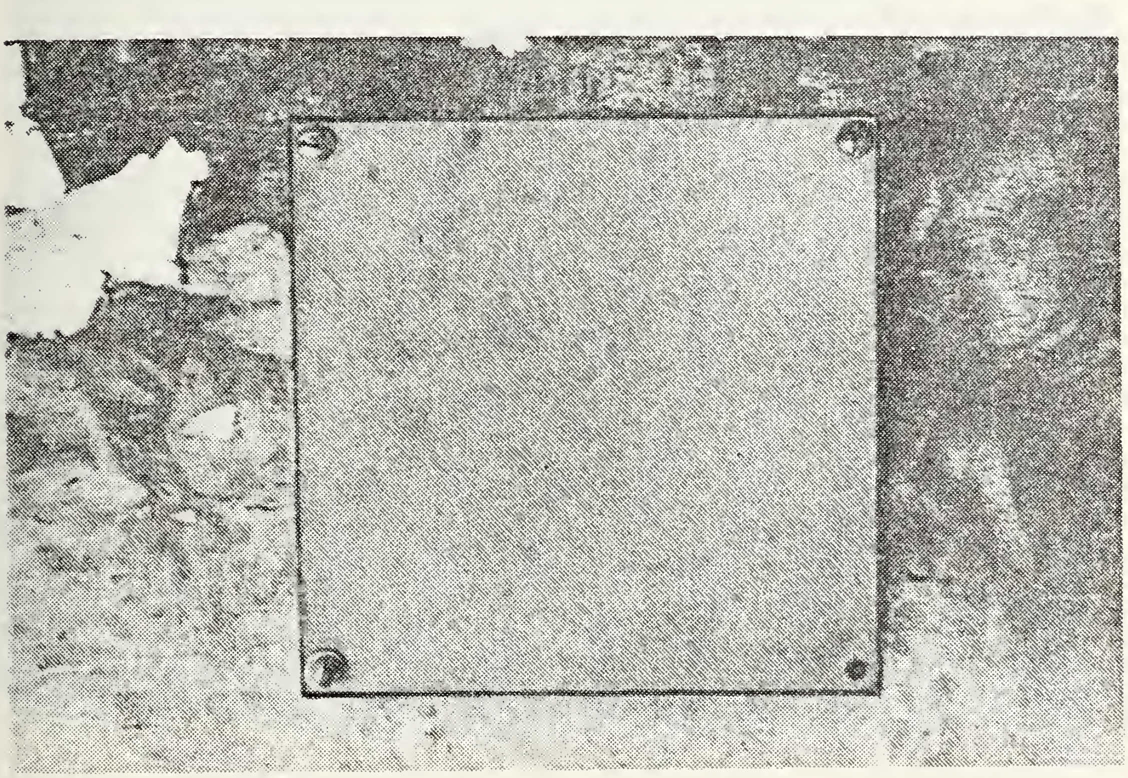


FIGURE 11: COMPOSITE PRIOR TO TESTING MOUNTED IN WINGBOX

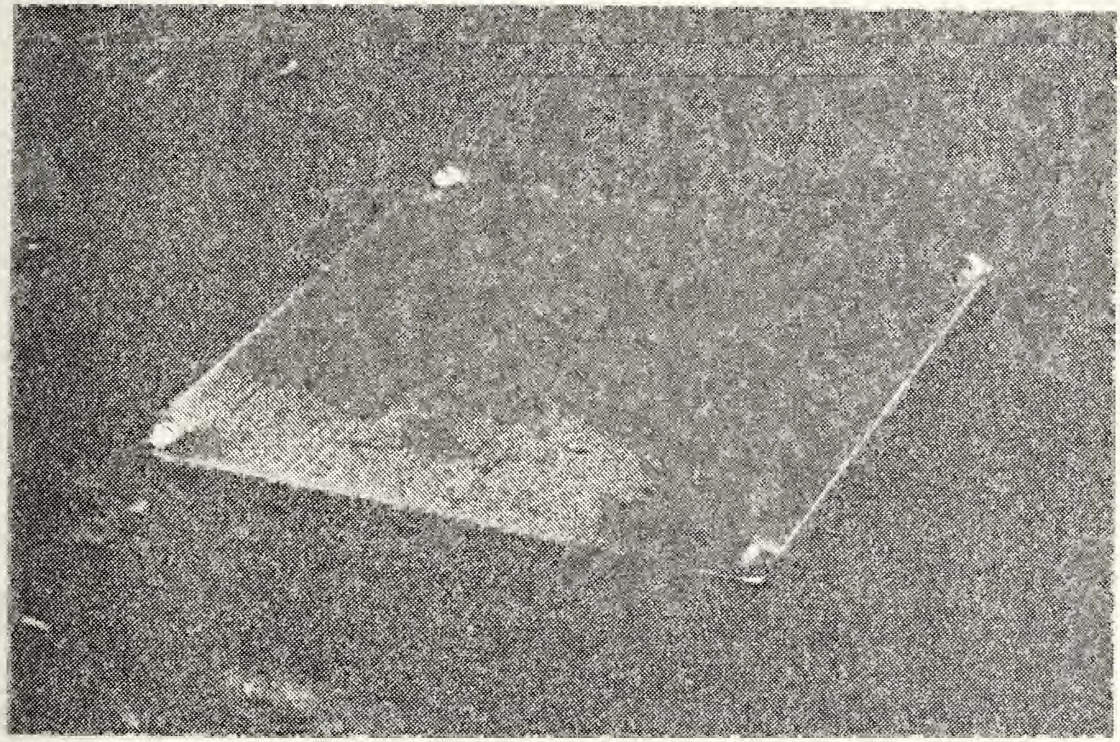


FIGURE 12: COMPOSITE AFTER JET BLAST TEST. NOTE CARBON BUILDUP

D. TESTING OF SAMPLES BY SHORT BEAM SHEAR TEST

The samples were sent to the Naval Postgraduate School for testing by the short beam shear test. The samples were all cut to the sizes specified in Table III. The ASTM standard test D 2344 was followed. A copy of this test is contained in Appendix C. The basic test procedure is as follows:

1. Cut the specimen(s) to the appropriate sizes. (Figure 2, Table III).
2. Measure and record the thickness, width and length of the specimen(s).
3. Turn on the INSTRON to allow ample warmup time (30 minutes) (Figure 18).
4. Set up the compression load cell.
5. Set the scale of the chart to 2 in./min. and maximum load to 500 lbf, 1000 lbf or 2000 lbf for 1/8, 1/4, 1/2 inch specimen thicknesses respectively.
6. The crosshead speed is then set to 0.5 in./min.
7. Set the appropriate test span on the specimen supports as per Table III (Figures 15, 16 and 17).
8. Center the specimen in the test fixture and align the midpoint to the center loading mechanism (Figure 17).
9. Apply the load to the specimen at the specified crosshead rate. Record the load to break the specimen.
10. Repeat 1, 2, 7, 8, and 9 for each specimen.

NOTE: The short beam shear test is not recommended for samples greater than 1/4 inch. This is the reason the shear strength drops off by approximately 15% for the 1/2 inch samples. The test was used for these samples as it was desired to obtain qualitative results for comparison purposes and not the exact shear strength of the sample. The test is adequate for these purposes as it gives consistent results even though they are low.

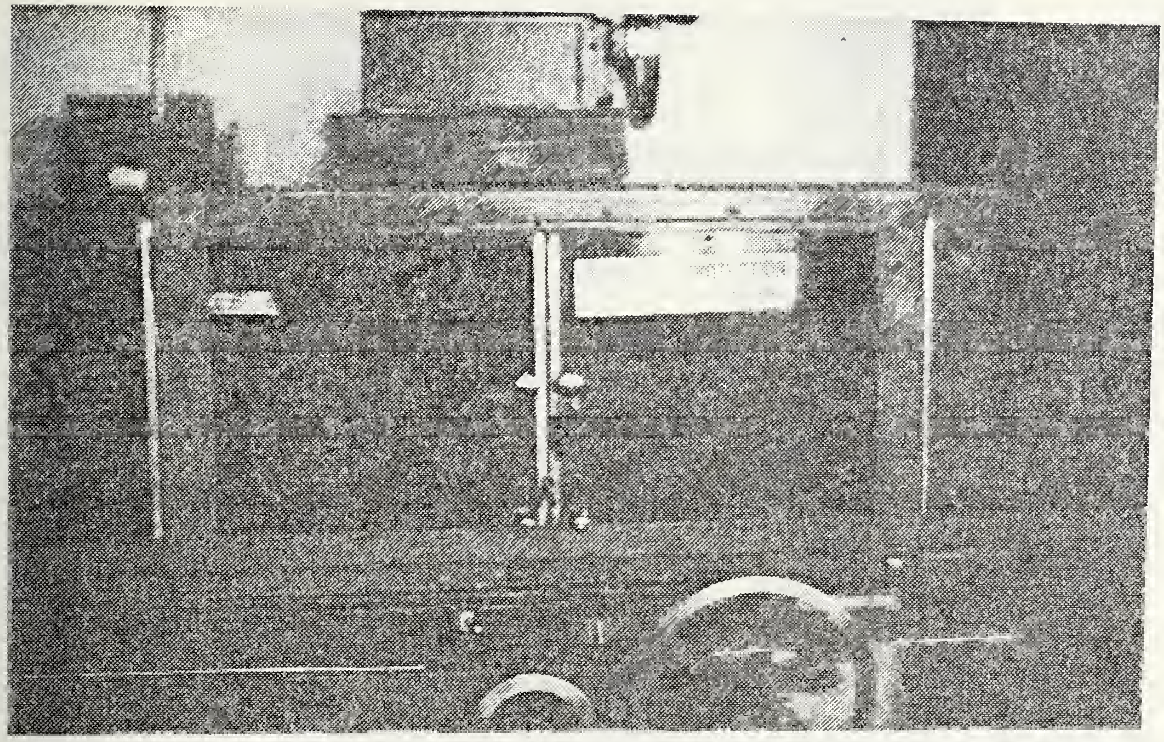


FIGURE 13: CUTTING MACHINE USED FOR LARGE CUTS AND ALL CUTS ON 1/2 INCH SAMPLE

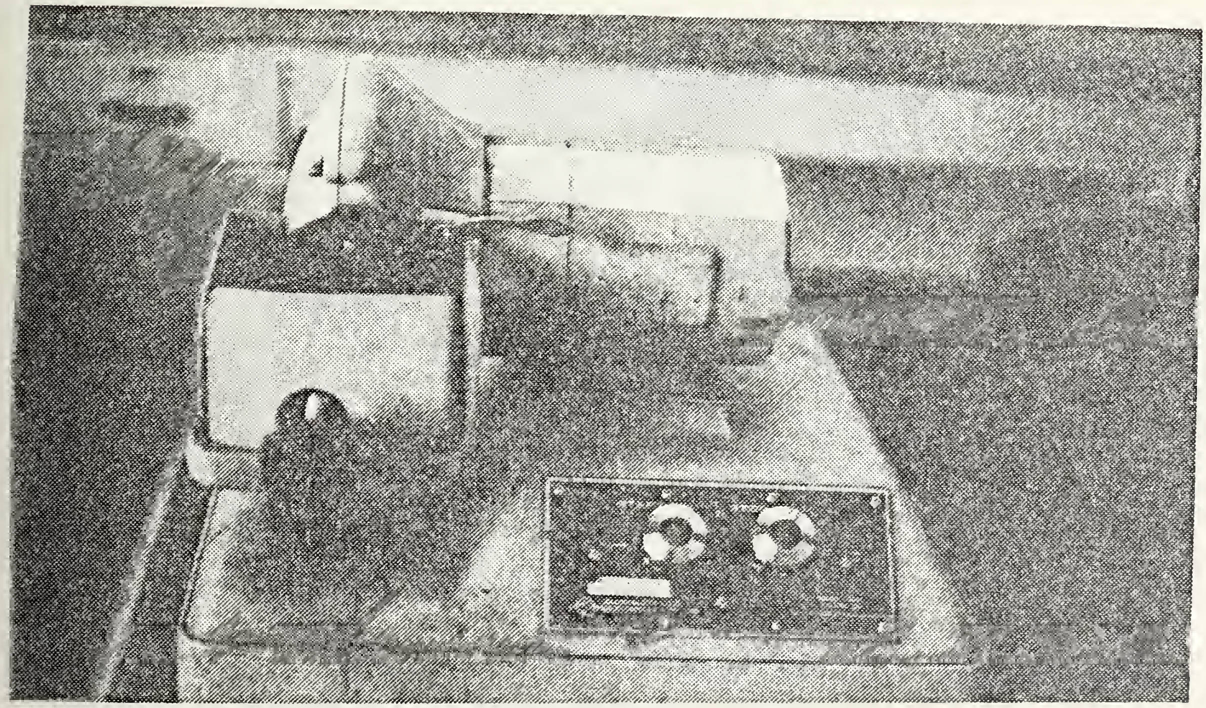


FIGURE 14: CUTTING MACHINES USED FOR FINISHING CUTS ON 1/4 AND 1/8 INCH SAMPLE

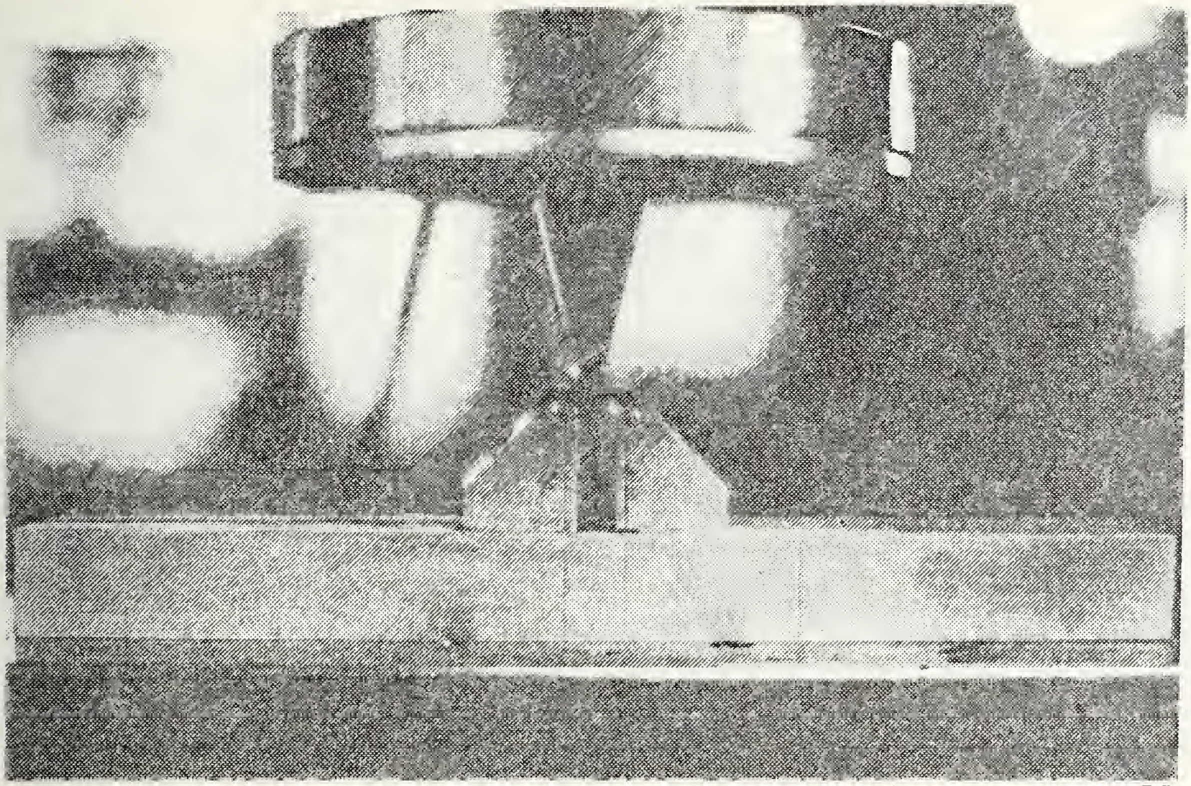


FIGURE 15: TEST RIG SETUP FOR TESTING 1/8 INCH SAMPLES

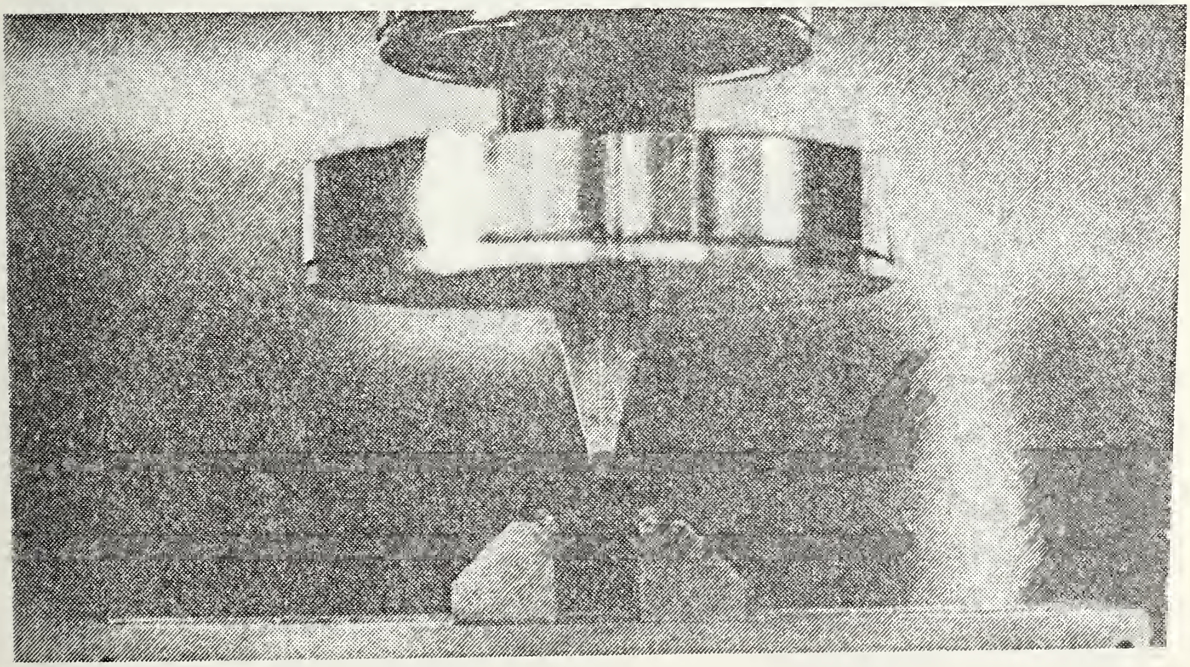


FIGURE 16: TEST RIG SETUP FOR TESTING 1/4 INCH SAMPLES

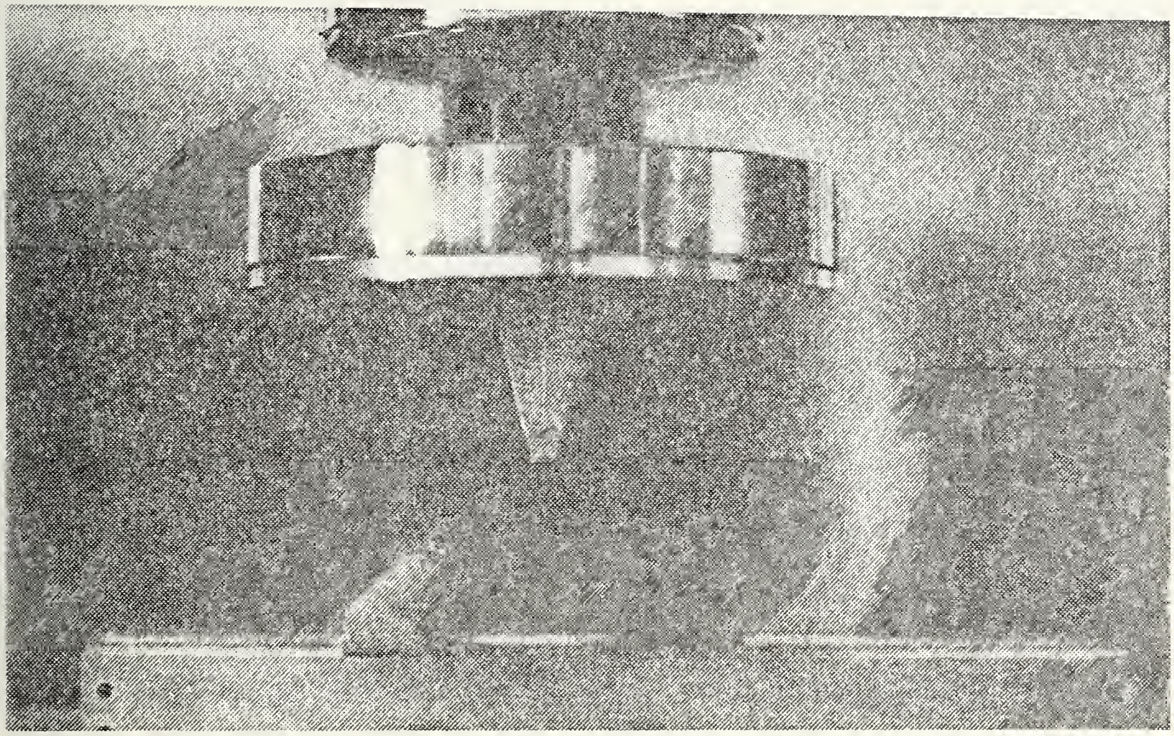


FIGURE 17: TEST RIG SETUP FOR TESTING OF 1/2 INCH SAMPLES

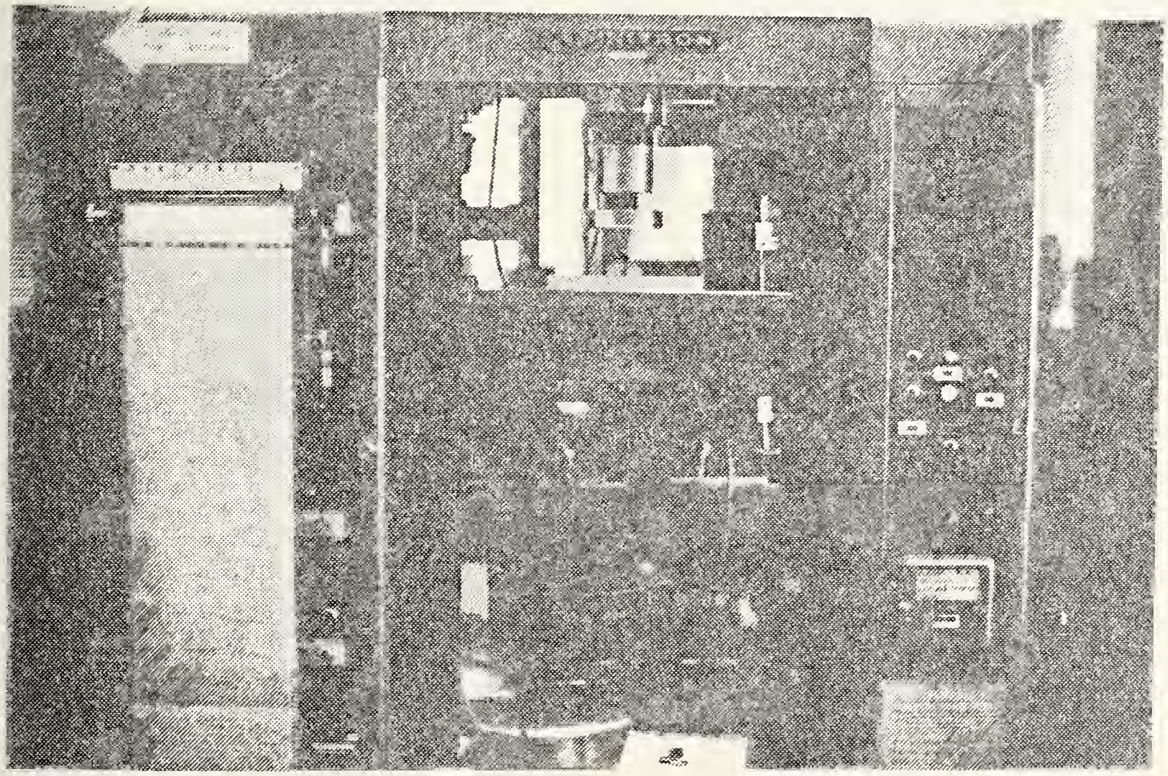


FIGURE 18: INSTRON SETUP FOR SHORT BEAM SHEAR TEST

IV. DATA

The raw data taken in the short bean shear test is included as Appendix D. Appendix D also includes the calculated shear stresses for each sample along with the mean values and standard deviations for each test.

A sample calculation for the shear stress is shown on the following page. The average of each test run are presented in Table IV and the percentage of original strength remaining after each test is included in Table V. An explanation of the objective of each test follows.

The original strength was taken to be the average of the two values presented by Hitco. This value was used as it was either equal to or greater than the values obtained at the Naval Postgraduate School with the untreated specimens, i.e., a conservative approach was taken. Hitco's results were also used, since their results are based on samples from the two edges of the plate and the location in the plate of our samples is unknown. This is due to the difficulty in manufacturing procedures and should be reduced in time as the manufacturing processes are refined.

A. SHEAR STRESS SAMPLE CALCULATION

The shear stress is calculated as follows:

$$S_H = 0.75\ P_B/bd \qquad (6)$$

S_H ≡ shear strength (psi or N/m^2)

P_B ≡ breaking load (lbf or N)

b ≡ width of specimen (in or m)

d ≡ thickness of specimen (in or m)

Sample calculation: Specimen 1A

$S_H = 0.75(410)/(.257)(.134) = 8929.08$ psi

NOTE: The thickness of the specimens in the data is with the paint on the specimen. The shear strength is calculated by substracting the paint thickness as it contributes nothing to the strength. The paint thickness was determined by measuring. the thickness of the composite in various areas and scraping off the paint and measuring the thickness in the same areas.

TABLE IV SHEAR STRESS AVERAGES			
TEST NUMBER	1/8" SAMPLES	1/4" SAMPLES	1/2" SAMPLES
1 (UNTREATED	9830 (AVG.TEST)	10030	8880
2 OVEN TESTS	N.A.	9030	N.A.
3 OVEN TESTS	N.A.	9430	N.A.
4 OVEN TESTS	N.A.	1650	N.A.
5 J	10300	10560	9340
6 E T	10200	10370	5090
7 T	9660	10540	9600
8 E E	10230	10850	9030
9 N	10200	10380	8880
10 G S	11330	10080	9260
11 I	10370	10610	8490
12 N T	10260	10180	8880
13 E	11430	10620	8630
HITCO'S AVG. UNTESTED	10440	10410	8860

TABLE V

PERCENTAGE OF ORIGINAL STRENTGH

TEST NUMBER	]/8 " SAMPLES	1/4 " SAMPLES	1/2 " SAMPLES
1	94	96	100
2	N.A.	86	N.A.
3	N.A.	90	N.A.
4	N.A.	16	N.A.
5	99	101	105
6	98	100	57
7	93	101	112
8	98	104	102
9	98	100	100
10	108	97	104
11	99	102	96
12	98	98	100
13	109	102	97
14 AVERAGE STD. DEVIA.	$\pm$ 8	$\pm$ 8	$\pm$ 6

B. TEST 1

Test number 1 is the results obtained in untreated specimens. The purpose of this test is to compare the results of Naval Postgraduate School tests with those conducted by Hitco at manufacture. Some of the samples from Hitco were not labeled as to whether they came from the right side of the plate or the left (significant as shear strength ranged from 10090 left side to 10790 right side for 1/8 inch thick plate), an average of the two sides were used. This is the value recorded on the last row of Table IV. Hitco's values were also used in computing the percent of original strength remaining.

Two 1/8 inch samples were tested The average of the Naval Postgraduate School tests was 9830 psi or 94% of Hitco's value. The 1/2 inch sample failed at 8880 psi or 100% of Hitco's value. These results were not as close to Hitco's as hoped but they were all within one standard deviation of Hitco's value. The large variance in values is due to the variation of the strength of the composite resulting from fabrication and not the testing procedure. This is based on the range of values obtained from the left side to the right side of the plate. The variation in strength is due to the variation in pressure or temperature over this plate during the cure cycle. It is very difficult to maintain a uniform temperature and pressure over the entire plate as the original plates are manufactured in large panel sections, 15 ft. by 15 ft.

The specimens all failed in shear along a 45° angle separation through the laminate and then following parallel to the laminate before proceeding at another 45° angle.

As mentioned previously, the paint thickness is taken into account in the calculation of the shear strength. The results of these tests show us the test results of Hitco can be accurately reproduced and general comparisons between untested and tested specimens may be made.

It was also decided that a change in shear stress was not to be considered significant unless it fell outside $\pm$ one standard deviation of Hitco's value. The criteria can be broken down into $\pm$ 8%, $\pm$ 8%, and $\pm$ 6% for the 1/8 inch, 1/4 inch and 1/2 inch samples respectively.

C. TEST 2

Test 2 considered samples cut from quarter inch nominal thickness plate. The plate was painted with white paint and subjected to a heat flux of 4.8 Btu/ft. sec.. The composite was removed from the oven when the temperature T_1 (1/16 inch from the surface of the plate) reached 204°C. The composite showed a slight discoloration in the paint near the center (the thermocouples were also mounted in this area) so the center section was cut out for the short beam shear tests. Eight samples were tested, with the result that the shear strength was 84% of the original shear strength. This test shows as expected, that the strength of the composite decreases when the temperature exceeds the cure temperature of 177°C.

Test specimen 2A was poorly cut and this value is discarded, the damage strength is 86% of the original shear strength.

The composite reached this temperature after approximately 1.5 minutes. The total heat absorbed would be 432 Btu/ft.

D. TEST 3

Test 3 samples were cut from quarter inch nominal thickness plate. The plate was painted with grey paint and subjected to a heat flux of 3.1 Btu/ft. sec. (2.7 $\frac{W}{cm^2}$). The composite was removed from the oven when the temperature T_1 approached 204°C. The composite showed no noticeable discoloration in the paint. The center section of this composite was selected for use in the short beam shear test as the thermocouples were mounted in the center so an accurate temperature was felt to be known. Eight samples were taken and tested with the result that the average shear strength was 90% of the original shear strength. This result is as expected as we have exceeded the cure temperature of 177°C.

The composite took about 2 minutes for T_1 to reach 200°C. The total heat absorbed by the composite is 744 Btu/ft.

E. TEST 4

Test 4 samples were cut from quarter inch nominal thickness plate. The plate was treated with white paint and subjected to a heat flux of 3.3 Btu/ft. sec. (2.9$\frac{W}{cm^2}$) for a period of 5 minutes. The temperature T_1 reached 325°C (617°F), which is significantly above the cure temperature. The sample

showed severely charred sections in the center of the plate along with delamination of the composite. The samples taken for testing were cut from the center section as this section was charred the worst and had the greatest amount of delamination. Eight samples were cut and tested with the result that the shear strength was 16% of the original values. Although a decrease in the strength was expected this large reduction in strength was not anticipated. The total heat flux absorbed by the composite is 990 Btu/ft.

G. DISCUSSION

The results of the initial set of Tests 1-4 show that:

1. Duplication of test procedures is achievable with sufficient degree of accuracy.

2. Composite materials have a severe loss in strength with increasing temperatures.

3. The maximum temperature reached by the composite appears to be the critical element vice the total heat absorbed by the composite. This is based on the fact Test 2 and 3 were removed at the same temperature but Test 3 received approximately 1.7 times as much heat and the strengths were within one standard deviation of each other.

4. A failure criteria was decided to be the point at which T_1 thermocouple exceeds 200°C. This was used as it gave us a strength reduction of 10 - 15% for the oven specimens.

It is now necessary to test the composites with actual jet engines to determine whether composites reach or exceed critical temperature at normal operating conditions.

H. JET BLAST TESTS

The procedure outlined on page   was used in subjecting the composites to the jet blast. The composite samples were mounted in the following order: 1/2 inch, 1/4 inch and 1/8 inch, with the 1/2 inch sample closest to the leading edge. Figure 6 shows the access for mounting the samples and Figure 11 shows a mounted sample.

The composites all had thermocouples mounted as per Table II. The tests were to be terminated when T_1 on the 1/2 inch specimen reached or exceeded 200°C. This temperature was selected because the oven tests at 200°C showed a 10 - 15% loss in strength.

The separation between the composites on the wingbox, was approximately 8 inches. It was assumed that due to the short span that all specimens were seeing approximately the same test conditions. The actual tests however, showed the 1/2 inch samples to run about 20°C higher than the 1/4 inch samples. The 1/4 inch and 1/8 inch samples however did not exhibit these trends. It was expected that the 1/2 inch samples should show the greatest changes in shear strength due to the higher temperatures indicated by the thermocouples.

1. Test Number 3

The first jet engine test was performed at engine idle. The test was to determine if the composite would reach or exceed critical temperature at steady state conditions.

The engine was set at 65% power level and allowed to stabilize. The wingbox was secured into place until the thermocouples reached a steady state condition. The thermocouples reached a steady state temperature after a 10 minute exposure. The wingbox was removed and the composites were allowed to cool down. At steady state, the maximum temperature obtained at the T_1 thermocouple of each sample was 125°C, 113°C and 108°C for the 1/2 inch, 1/4 inch, and 1/8 inch samples respectively.

The short beam shear test of these samples showed no loss in strength. The strengths of these samples were 105%, 101%, and 99% of the original strength for the 1/2 inch, 1/4 inch, and 1/8 inch samples respectively. This was as expected as the sample temperatures remained below the cure temperature. The slight increase is easily explained by the fact that additional curing could take place, during exposure to a thermal environment, relieving some of the residual stresses introduced during the curing cycle.

The conclusion reached from this test is that the samples will not reach a critical temperature and degrade at engine idle conditions. The fact the temperatures did not approach critical values permits the elimination of the idle engine power settings from further consideration.

2. Test Number 6

This test was designed to determine the engine power setting which would result in the composite temperatures which reached or exceeded the critical temperature at steady state. For an 80% power setting, the composite temperatures reached a steady state value of 150°C at the four minute mark of the test. Since this temperature was well below the critical temperature, the power setting was increased to 90% after 3.5 minutes at the 90% power setting the thermocouples behaved erratically, and the wingbox was removed from the jet blast. It was determined that the erratic readings were caused by a crack which developed at the leading edge of the wingbox. This crack allowed the exhaust gases inside the wingbox causing the thermocouple cabling to fuse together. The maximum temperatures obtained at the T_1 thermocouples (before the erratic readings) were 220°C, 237°C, and 226°C for the 1/2 inch, 1/4 inch, and 1/8 inch samples respectively. It is expected that the thermocouple on the 1/2 inch sample failed first. This is due to the fact the 1/4 inch reading was usually 20°C lower than the 1/2 inch sample. Therefore, it is estimated that the temperature of the 1/2 inch sample was in excess of 237°C.

The short beam shear tests showed a negligible change of strength with the exception of the 1/2 inch sample. The samples had strengths of 57%, 100%, and 98% of the original values for the 1/2 inch, 1/4 inch, and 1/8 inch samples respectively.

Due to the oven tests, it was expected that decreases in strength for these samples greater than 15% would result. The fact that the 1/4 inch and 1/8 inch samples showed no change in strength, was unexpected. The temperature reached in these samples were in excess of both the cure and the critical temperature.

The 1/2 inch sample showed separation and delamination at the leading edge (Figure 19). It was therefore expected that the sample would show a decrease in strength. The 43% loss in strength was more severe than expected. This significant change in strength also appears to point out that the temperature was in excess of the 220°C recorded. This is due to the fact the other samples exceeded this temperature (according to the thermocouples), but showed no loss in strength.

It was necessary at this time, due to the damage to the wingbox and the wiring to replace the damaged wiring and reweld the wingbox prior to any further testing.

3. Test Number 7

Test number 7 was designed to determine the power setting at which critical temperatures could be reached at steady state. The last test showed that an 80% power setting resulted in temperatures below the critical temperature, and the 90% power setting exceeded the critical temperature at steady state. The engine power was started at 82% and increased to 84% and then 86% after steady state was reached

at each power setting. The maximum temperature reached by all the samples for the 82% power setting was 166°C after a 5 minute exposure. The power was then increased to 84% and a steady state temperature of 206°C was obtained at the T_3 thermocouple on the 1/2 inch sample. This steady state temperature was reached after 3 minutes of exposure. The power was then increased to 86% and steady state temperatures were reached after a 5 minute exposure.

The maximum temperatures obtained were 232°C, 196°C and 185°C in the 1/2 inch, 1/4 inch, and 1/8 inch composites respectively. Visual inspection of these composites showed no damage. The shear stress showed a slight increase in strength for the 1/4 inch sample being 101% of its original value (negligible change well within the standard deviation of $\pm$ 8%).

The 1/8 inch sample and the 1/2 inch samples showed conflicting results, the former decreasing in strength, the latter increasing. The 1/8 inch was 93% of the original value vice 112% of the original value of the 1/2 inch sample. The 1/8 inch sample is not considered significant as it is within one standard deviation. The 1/2 inch sample is significant for two reasons: 1) the strength increase is outside the one standard deviation (almost two standard deviations); 2) the result seems to contradict the result of Test 6, which showed a considerable decrease in strength.

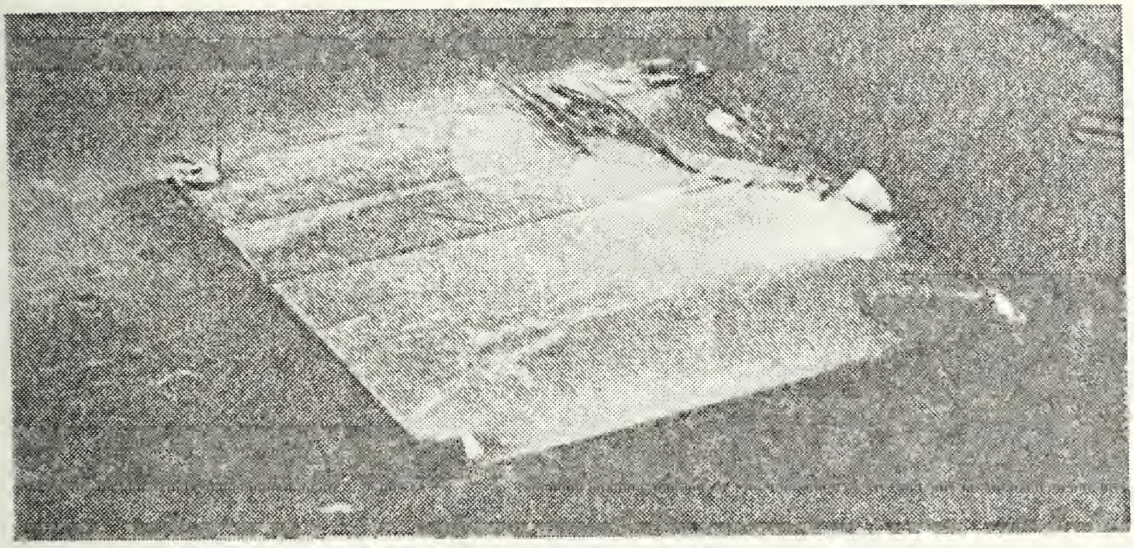


FIGURE 19: 1/2 COMPOSITE AFTER TESTING. NOTE DELAMINATION
TEST NUMBER 6

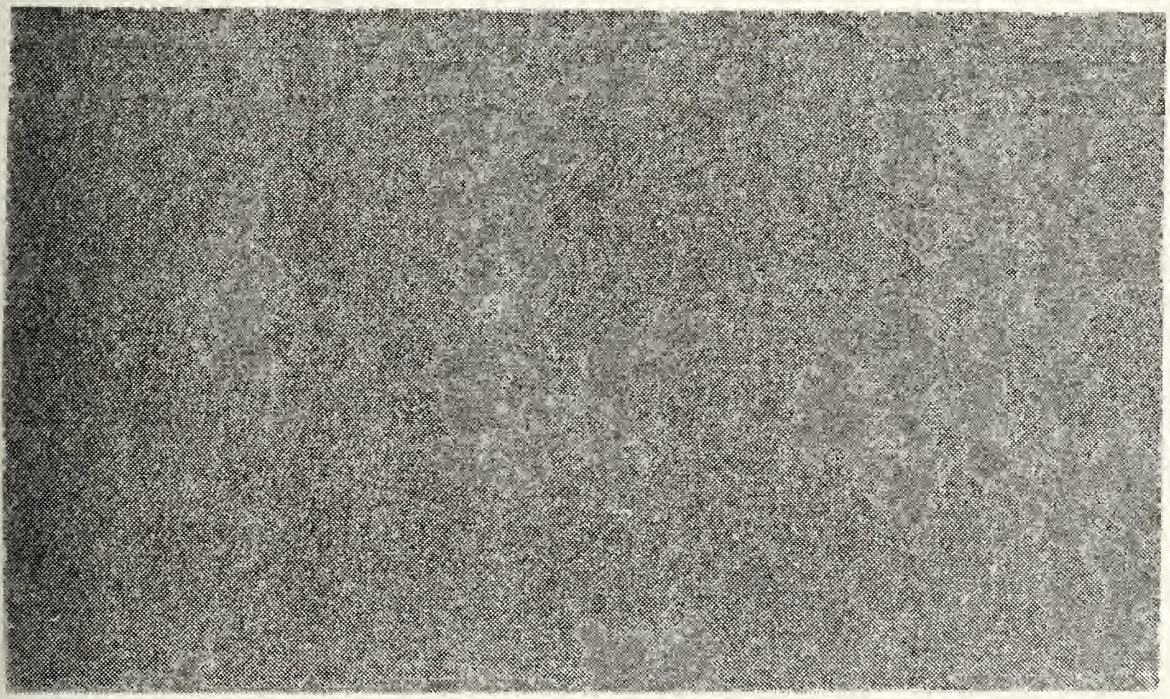


FIGURE 20: 1/4 COMPOSITE AFTER TEST 11 (8X MAGNIFICATION)
NOTE DELAMINATION IN CENTER

The one possible explanation for the contradiction is that the temperature in Test 6 far exceeded the recorded value of 230°C, as the temperature in this test was 230°C. It was noted that the temperature in Test 6 was far in excess of that recorded and this also seems to verify that belief.

4. Test Number 8

Test number 8 showed that an 86% power setting would in the steady state condition exceed the critical temperature of 200°C. Test number 8 was then designed to determine if cyclic effects cause progressive deterioration of the composite.

The test sequence was to expose the composite to the 86% power setting till the T_1 thermocouple in the 1/2 inch sample reached a temperature of 205°C. The sample was removed till T_1 cooled down to approximately 165°C. This sequence was to be repeated five times. The heating portion of each cycle took 5 minutes and 30 seconds and the cool down cycle took 7 minutes.

The specimens showed no visual damage after the testing. The shear stress tests showed no significant strength changes. The samples were 102%, 104%, and 98% of the original values for the 1/2 inch, 1/4 inch, and 1/8 inch samples respectively. These results again were not anticipated as the temperatures exceeded the critical temperature. It was expected to show a 10 - 15% decrease in strength. This result seems to justify the belief that the temperature is not the controlling

factor unless it is extremely high (i.e. greater than 250°C). It appears additional curing may be taking place.

5. Test Number 9

Test number 9 was designed to show the effects of absorbed moisture on the composites. The samples were soaked in salt water for seven days prior to testing. The percent gain in weight due to moisture absorption was 2%, 3%, and 5% for the 1/2 inch, 1/4 inch, and 1/8 inch specimens respectively.

The test was conducted at 86% power setting with the samples removed at a temperature of 200°C. It took a significantly longer time to heat these samples to 200°C than the unsoaked samples. (8 minutes 40 seconds vice 5 minutes 30 seconds). This was attributed to the fact the water was vaporizing in certain areas of the plate thus carrying away some heat from the composite.

The shear test again showed no change in strength. The strengths are 100%, 100% and 98% of the original strength for the 1/2 inch, 1/4 inch, and 1/8 inch samples respectively. The samples had no visual damage after exposure to the jet blast.

6. Test Number 10

Test number 10 was an 86% power setting. The samples were removed after a temperature at T_1 of 200°C was reached. The samples took 3 minutes and 10 seconds to reach this temperature. Visual inspection showed no damage to the composites.

The 1/2 inch and 1/8 inch samples showed slight increases in strength having 104% and 108% of the original strength respectively. The 1/4 inch sample showed a slight decrease maintaining 97% of its original strength. There was no visual damage to the specimens.

7. Test Number 11

Test number 11 was a three cycle test. The power setting was 86%. The samples were removed when temperature T_1 reached 205°C and reinserted into the gas flow when the temperature dropped to 180°C. It is significant to note that it took approximately the same time to heat to 205°C as Test 8 where the samples were cooled to 165°C. This shows that the temperature rise is nonliniar.

The 1/4 inch sample showed some delamination (Figure 20) in a small section of the composite. The other samples showed no visual deterioration. The shear strength of the 1/4 inch specimen actually showed a slight increase in strength being 102% of the original strength. The 1/8 inch and 1/2 inch samples showed slight decreases in strength being 99% and 96% respectively. It was surprising that the 1/4 inch sample showed a slight increase in strength. It was expected to have deteriorated due to the visual observations. The delamination observed in the composite was limited to a section .25" by .25", but it was thought that degradation would occur before becoming visible.

It therefore seems from this test that isolated delamination does not serously affect the overall strength of the composite.

8. Test Number 12

This test was conducted at 89% power. The sample remained in the jet blast to a temperature T_1 of 210°C. The time necessary to reach this temperature was 90 seconds.

Visual inspection of the samples showed not changes in the specimens. The shear test showed slight decreases in strength for both the 1/8 inch and 1/4 inch samples. The samples had a strength of 98% of the original. The 1/2 inch specimen had the same shear strength as the original samples.

9. Test Number 13

The last test was conducted at 89% power setting. The test was designed to reach a temperature at T_1 of 220°C to duplicate test number 6 with the exception that the power was reduced to 89% vice 90%. The test was terminated when T_1 on the 1/2 inch sample reached 220°C. The time to reach this temperature was 3 minutes 30 seconds. The composites were removed from the jet exhaust blast and tested by the short beam shear test.

The 1/4 inch and 1/8 inch samples showed increases in strength to 102% and 109% respectively. The 1/2 inch specimen showed a slight decrease in strength maintaining 97% of its original strength.

This test seemed to verify that the temperatures in Test 6 definately exceeded the value of 220°C in T_1 of the 1/2 inch sample. As it seems highly unlikely to have such a large discrepancy in material degradation for the same temperature.

The 1/4 inch sample showed slight delamination of the composite in a very small area in one dorner (Figure 21). The results of the shear test for the 1/4 inch sample indicates that small areas of delamination do not imply a degradation of the entire composite.

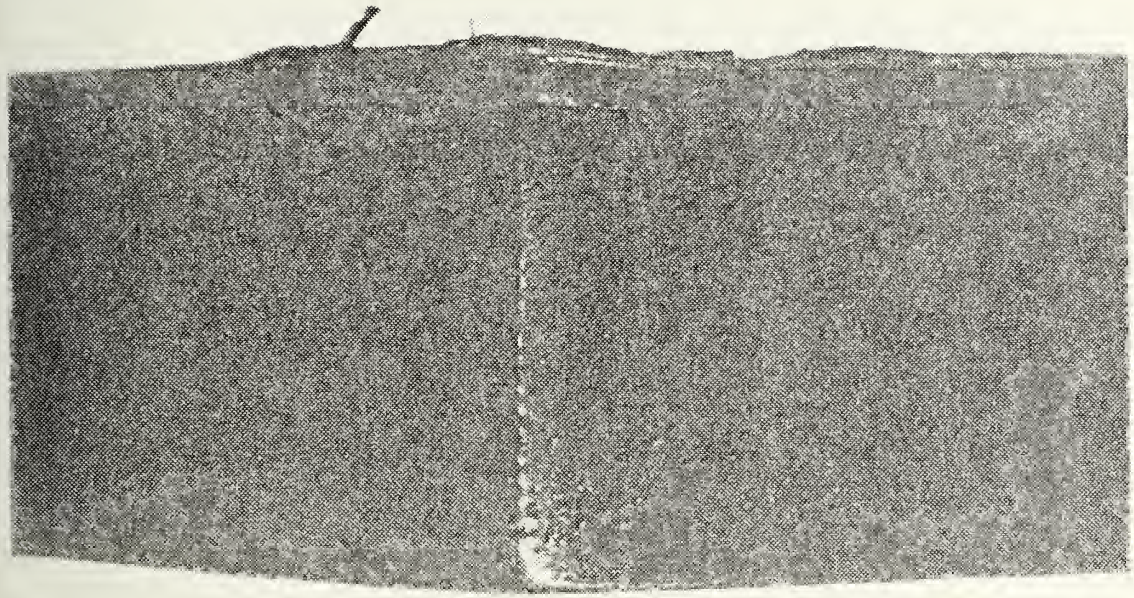


FIGURE 21: 1/4 COMPOSITE AFTER TEST 13 (10X MAGNIFICATION). NOTE DELAMINATION AT CORNER

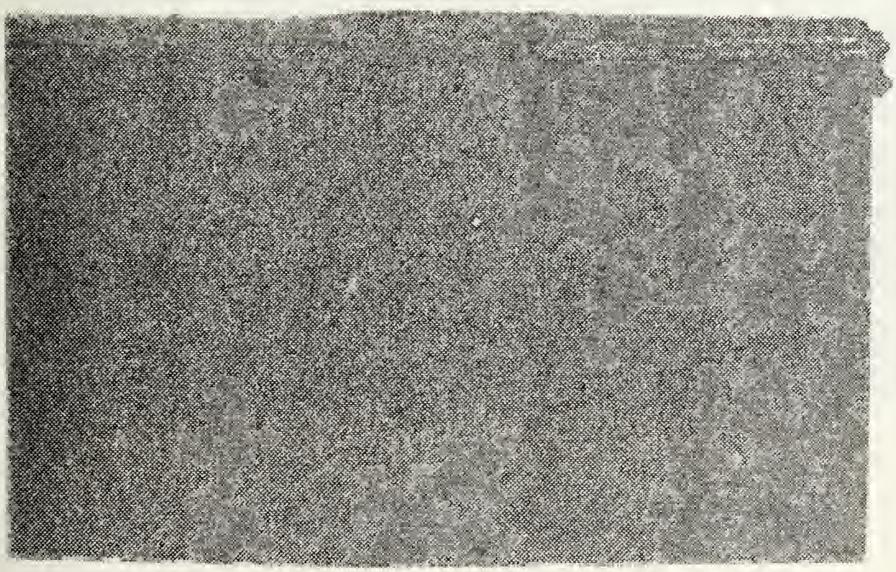


FIGURE 22: SAME AS ABOVE (12X MAGNIFICATION)

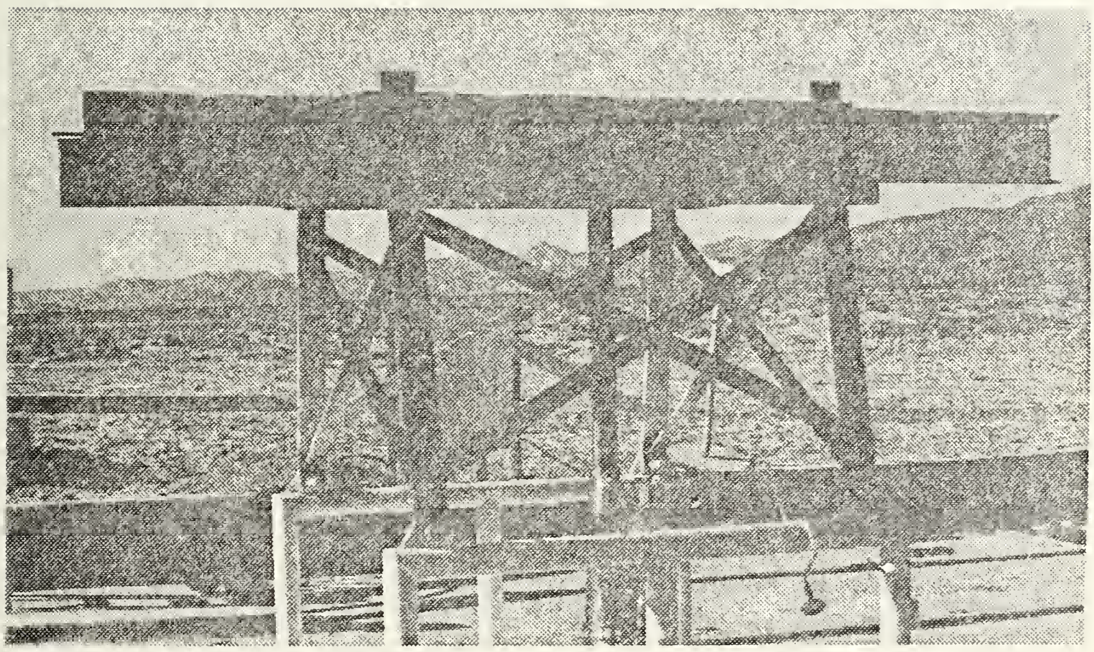


FIGURE 23: WINGBOX AT CONCLUSION OF TESTING. NOTE SOOT BUILD-UP ON WINGBOX

V. CONCLUSION

The standard deviations in strength for the tests varied from 390 - 1320, 640 - 1210 and 330 - 1370 psi for the 1/8 inch, 1/4 inch and 1/2 inch samples respectively. Taking into account the average standard deviation it was thought that a change in strength was significant if it was outside this range. The range being ± 8%, ±8%, and ± 6% of the original strength for 1/8 inch, 1/4 inch, and 1/2 inch samples respectively.

The 1/8 inch samples had two results outside of the above mentioned ranged (Test 10 and 13). Both of these tests showed an increase in strength. The maximum temperature was 183°C for Test 10 and 206°C for Test 13. The results on the 1/8 inch specimens therefore, showed no significant degradation due to the jet exhaust blasts. The critical temperature determined from oven tests (200°C), however, was only reached or exceeded in three of the nine cases. These tests all exceeded the cure temperature of the composite (177°C) which previous tests had shown [Refs. 2, 3] to be the start of material degradation.

The 1/4 inch samples had no results outside the range of ± 8% of the original strength. All the results were within ± 4% of the original strength. Therefore, it can be concluded that there was no measureable degradation in the composites in any of the jet blasts tests. All tests exceeded the cure

temperature and Tests 6, 11 and 13 exceeded the failure temperature of 200°C determined by the oven tests.

The 1/2 inch specimens had two results outside the range of $\pm$ 6% of the original strength (Test 6 and Test 7). The shear strengths from Test 6 was significantly lower than the original strength, with only 57% of the untested sample strengths. The temperatures from this test are inaccurate due to thermocouple failure. It is only known for certain that the temperature exceeded 220°C. The thermocouple failed at that temperature and the composite remained in the blast for about another 45 seconds. The temperatures were still increasing at thermocouple failure so it is certain the temperature exceeded the recorded value of 220°C. This is further verified by the fact that the thermocouple wires had fused together and the insulation on the thermocouples was rated for 370°C. It is unlikely that the temperatures reached that value due to the short duration of exposure after thermocouple failure, but the temperatures were undeniably in excess of 220°C. This sample also showed visible signs of delamination (Figure 19). Test 7 reached a maximum temperature of 230°C. Test 7 sample however, showed an increase in strength to 112% of the original strength. Six of the nine jet blast tests exceeded the 200°C limit, two as discussed above, and of the other four tests, two were slightly above the original strength and two slightly below but all results were within a $\pm$ 5% range.

The one general conclusion that can be reached is that severe degradation of the composite properties can be visually detected. A visual inspection of the composite should concentrate on discoloration of the paint and delamination. Isolated sections which show these effects if less than 1 square inch, should maintain most of the original strength. If the discoloration or delamination covers an area greater than 1 square inch then significant damage is likely to have occurred.

It also appears that heating methods may have a significantly greater effect on the composites than the final temperature reached. This is based on limited data due to the fact only three tests were conducted in the oven. The different methods of heating are radiation for the oven and forced convection for the jet exhaust. The References [2, 3], cited earlier which showed degradation occurring at temperatures above 177°C were heated in an oven. Thus degradation of composites should concentrate on the final temperatures obtained along with the way in which these temperatures were reached. It is unknown why the heating method has this effect or even if this is a true statement, but comparison of the various tests seems to point in that direction.

It is also concluded that under normal operating conditions aboard aircraft carriers, composites would not degrade to jet exhaust blast exposure. The conditions as set up in the original tests were worse case conditons that should seldom, if ever, be reached in actual operation. It would be possible

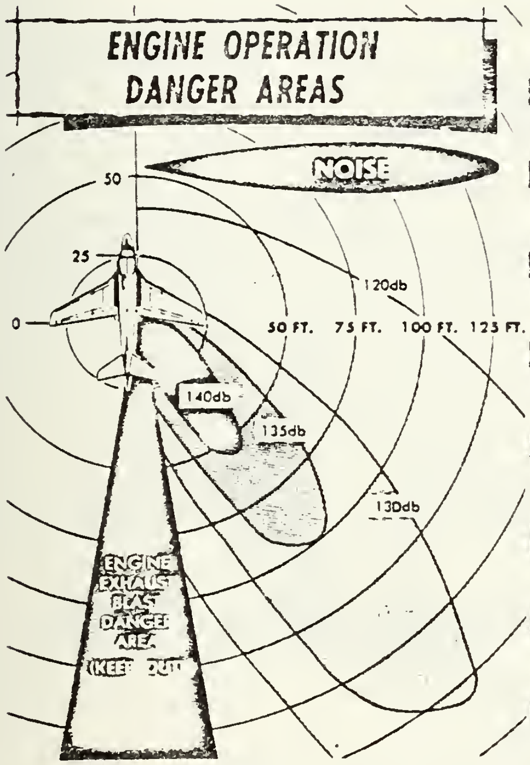
for the aircraft to be within 10 feet of another aircraft; but with the exception of the idle condition, the time duration and power level are unlikely. This is due to the fact that the F-14 only needs 80% power to start taxi, and idle power to maintain a taxiing condition [Ref. 13]. Therefore, the only time a jet should exceed 80% power settings is for taxiing or takeoff operations. During the time the plane is moving, the distance and duration will be increasing and decreasing respectivley.

VI. RECOMMENDATIONS

Further oven tests should be conducted to fill in the strength degradation between 200°C and 325°C. The composite load carrying sections of the F-18 should be visually examined prior to each flight to check for delamination and paint discoloration. If any of these effects are noted, further testing of the composites is required to ensure there is no severe strength loss.

Followon testing in this area should concentrate on the environmental effects on the composites over the life of the aircraft. This is due to the fact that absorbed moisture from the environment has a detrimental effect on the strength.

APPENDIX A: JET EXHAUST TEMPERATURE AND VELOCITY PROFILES



NOTE

NOISE DANGER AREAS SYMMETRICAL ABOUT ENGINE CENTERLINES DURING DUAL ENGINE OPERATION AT MAXIMUM POWER.

CAUTION

IF ENGINES ARE RUN UP IN FRONT OF BLAST DEFLECTOR, SOUND IS DEFLECTED TO SIDES RESULTING IN A DISTORTION OF PATTERN ILLUSTRATED.

DAMAGE RISK CRITERIA					
EAR PROTECTION NECESSARY	EXPOSURE TIME DURATION PER DAY				
	5 Minutes	15 Minutes	30 Minutes	1 Hour	2 Hours
NO PROTECTION	120db	115db	112db	109db	106db
EARPLUGS WITH AVERAGE SEAL	132db	127db	124db	121db	118db
EARPLUGS AND EARMUFFS	140db	135db	132db	129db	126db

REF. WADC TN 55-355 TO DAMAGE RISK CRITERIA

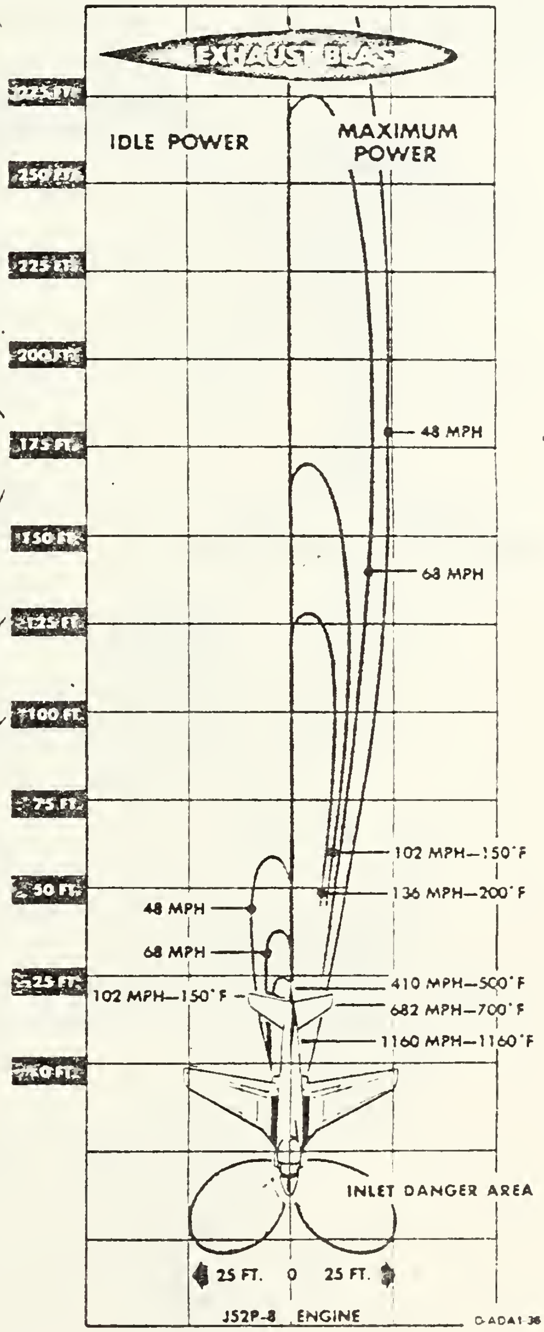
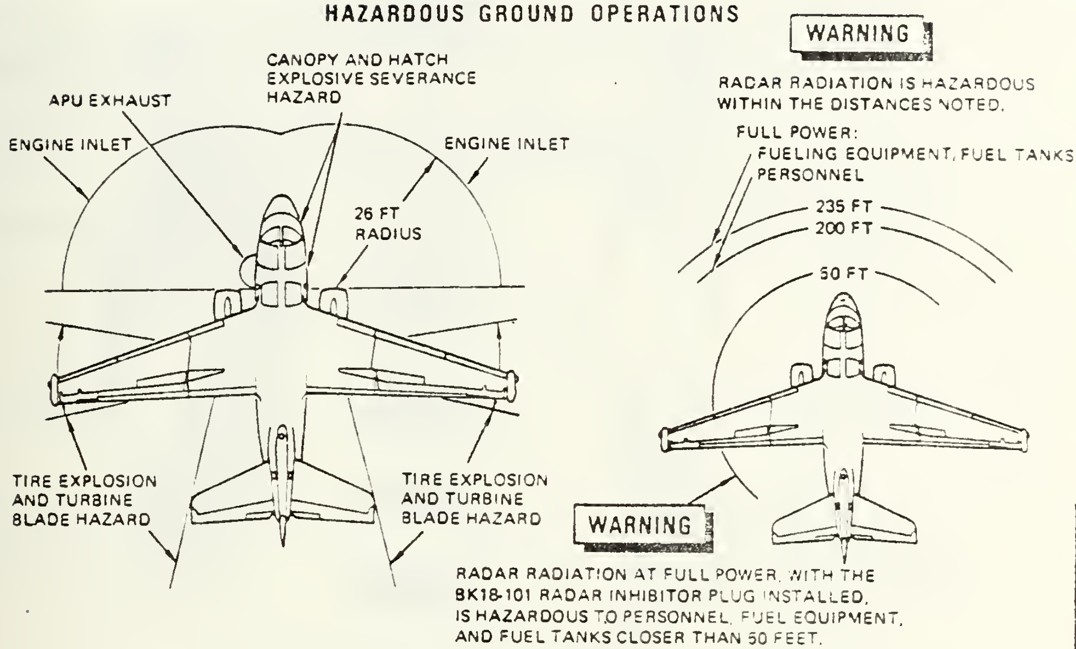


Figure 1-38. Engine Operation Danger Areas

A-6E/A-6E

DANGER AREAS

HAZARDOUS GROUND OPERATIONS



ENGINE BLAST AND TEMPERATURE AREAS

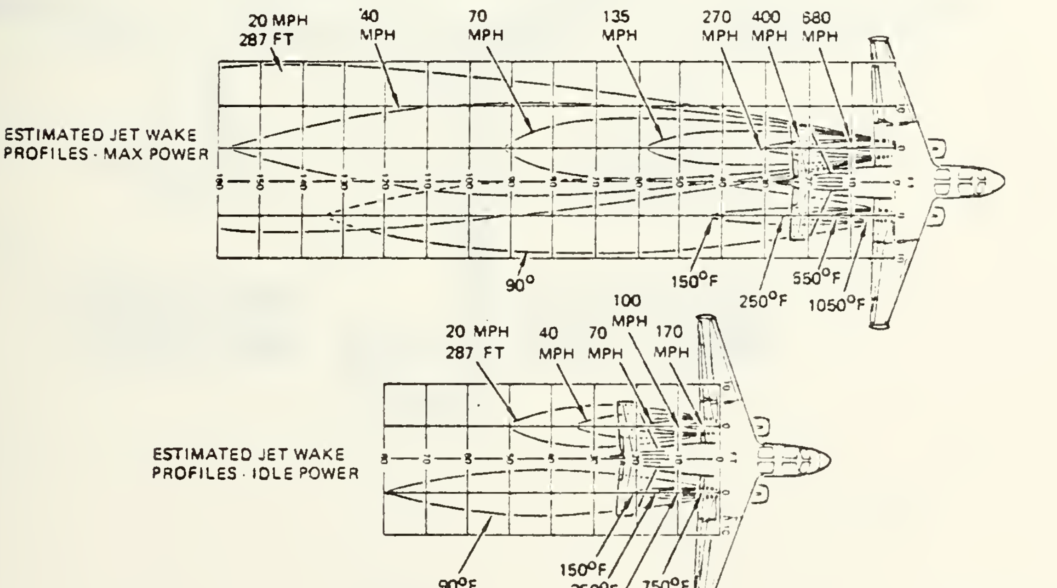


Figure 1-128 (Sheet 1 of 3)

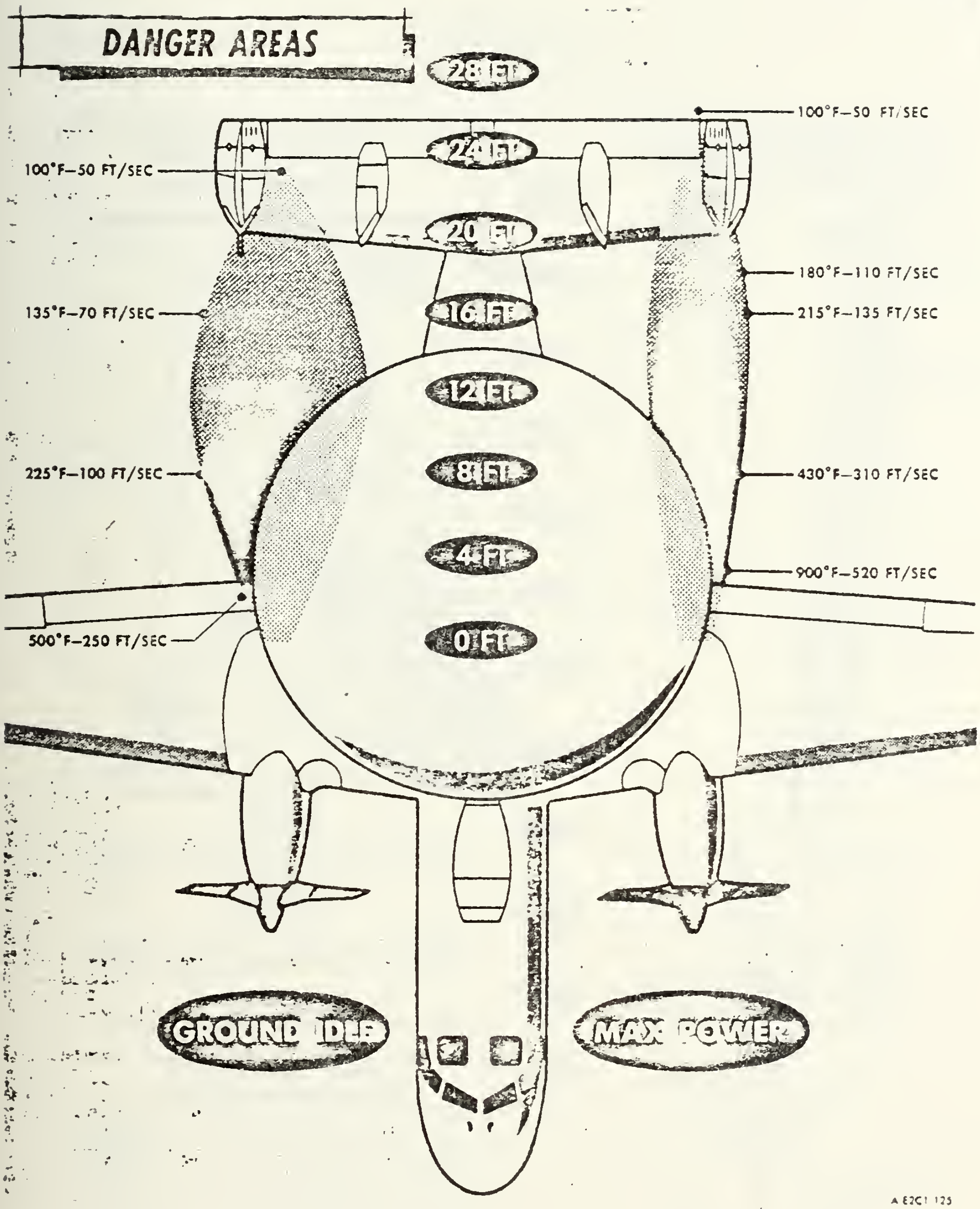
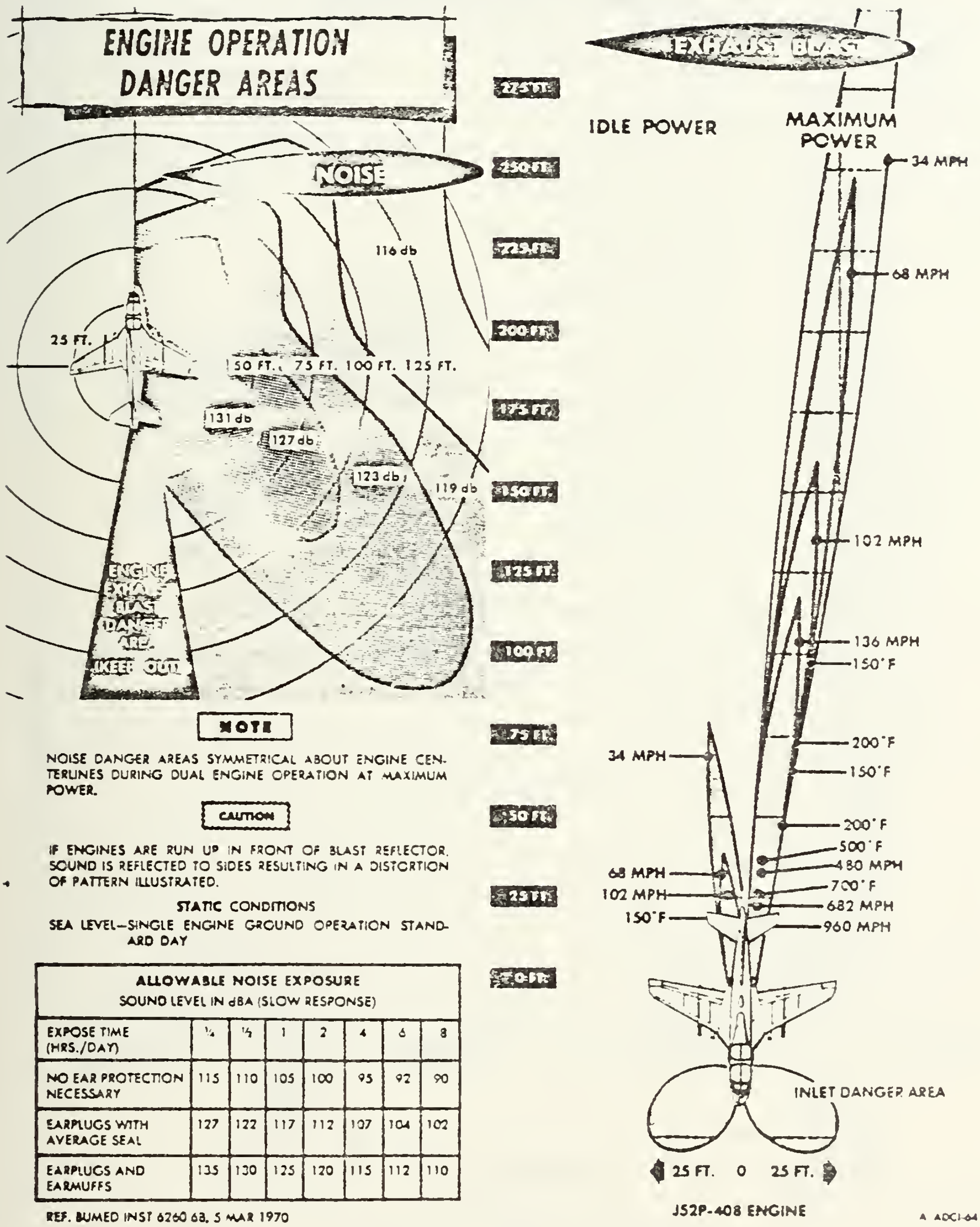


Figure 1-71

E-2C



NOTE

NOISE DANGER AREAS SYMMETRICAL ABOUT ENGINE CENTERLINES DURING DUAL ENGINE OPERATION AT MAXIMUM POWER.

CAUTION

IF ENGINES ARE RUN UP IN FRONT OF BLAST REFLECTOR, SOUND IS REFLECTED TO SIDES RESULTING IN A DISTORTION OF PATTERN ILLUSTRATED.

STATIC CONDITIONS

SEA LEVEL—SINGLE ENGINE GROUND OPERATION STANDARD DAY

ALLOWABLE NOISE EXPOSURE SOUND LEVEL IN dBA (SLOW RESPONSE)							
EXPOSE TIME (HRS./DAY)	¼	½	1	2	4	6	8
NO EAR PROTECTION NECESSARY	115	110	105	100	95	92	90
EARPLUGS WITH AVERAGE SEAL	127	122	117	112	107	104	102
EARPLUGS AND EARMUFFS	135	130	125	120	115	112	110

REF. BUMED INST 6260 6B, 5 MAR 1970

Figure 1-24. Engine Operation Danger Areas

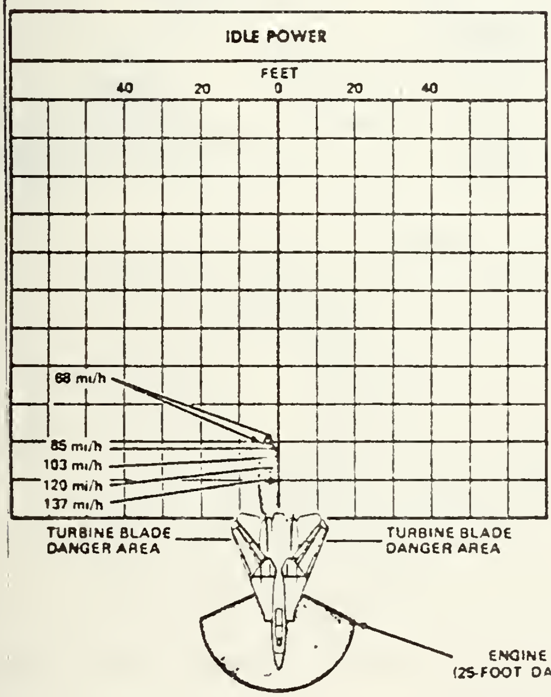
RUNUP DANGER AREAS

WARNING

- At high thrust settings, the danger area around engine inlets may extend as far as 4 feet aft of the inlet lip.
- Ear protection shall be worn at all times.

NOTE

- This illustration contains estimated temperature distribution of TF30-P-412A or P-414 afterburning turbofan engine per Pratt and Whitney Specification N-6131 with afterburners at maximum nozzle opening for idle power and maximum power (zone 5). Afterburner nozzles are fully closed (minimum opening) for military power.
- If engines are run up in front of blast deflector, exhaust jet wake is deflected up and to sides, resulting in distortion of patterns shown.



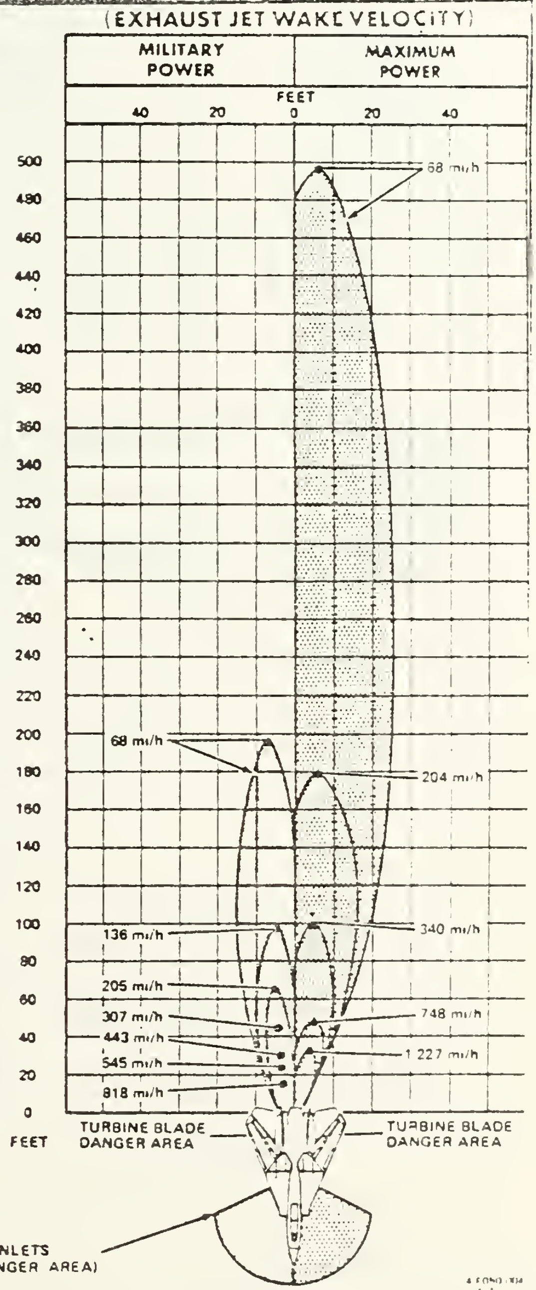


Figure 1-90. Runup Danger Areas (Sheet 2 of 2)

DANGER AREAS TF30-P-408 ENGINE

EXHAUST, INLET AND TURBINE DANGER AREAS

WARNING

Do not operate the AN/APQ-126(V) radar in congested areas. Personnel, explosives, and combustible materials must be kept out of the electromagnetic radiation hazardous area when the radar is transmitting. The safe distance for personnel is 20 feet. The safe distance for explosives is 30 feet. The safe distance for refueling operations is 45 feet.

WARNING

Remove all objects under the aircraft. Failure to do so can result in damage to aircraft or injury to personnel.

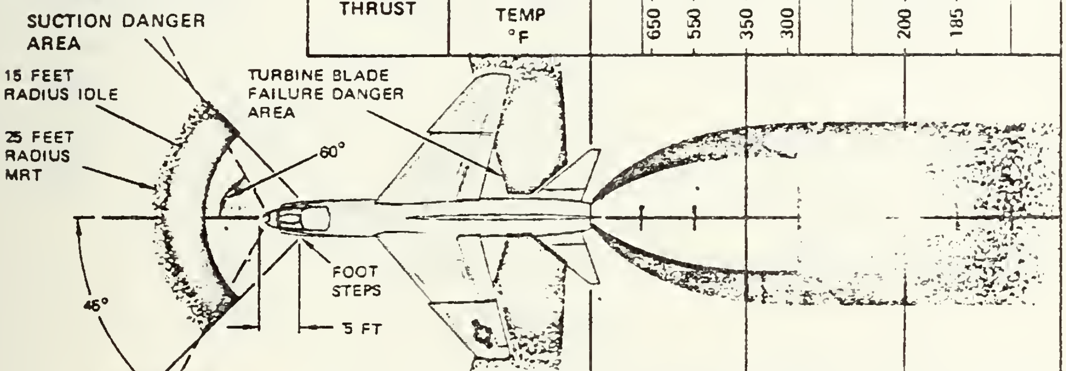
Stay clear of area within 100 feet directly behind the aircraft when the engine is operating at MRT.

LEGEND

- Military Thrust Danger Areas
- Idle Thrust Danger Areas
- Radiation Hazard

FEET 0 30 60 90

MILITARY THRUST	VELOCITY KNOTS	862	415	244	196	148	
	TEMP °F	650	550	350	300	200	185

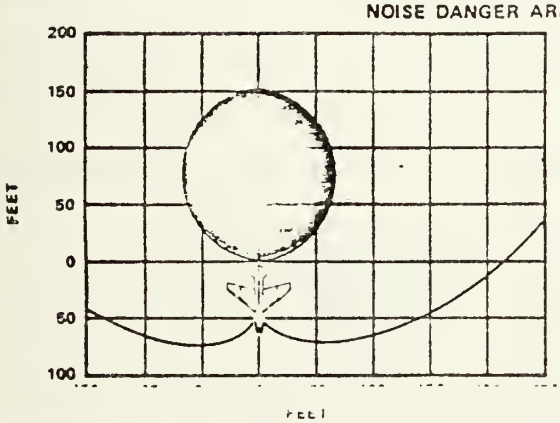


IDLE THRUST	TEMP °F	200	150			
	VELOCITY KNOTS	207	148	119	89	60

CAUTION

Make certain that suction danger area is clear of debris. Jet blast zone varies according to prevailing wind.

NOISE DANGER AREAS



- Military Thrust — 140 DB and above. Dangerous even with ear protection. Limit exposure to absolute minimum.
- Idle Thrust — 95 DB and above. Ear protection required.
- Military Thrust — 120 DB to 140 DB. Ear protection required. Use caution in exposure time.
- Military Thrust — 90 to 120 DB. Ear protection required.

Figure 1-73

A-7C A-7E

DANGER AREAS TF41-A-2 ENGINE

EXHAUST, INLET AND TURBINE DANGER AREAS

WARNING

Do not operate the AN/APQ-126(V) radar in congested areas. Personnel, explosives, and combustible materials must be keot out of the electromagnetic radiation hazardous area when the radar is transmitting. The safe distance for personnel is 20 feet. The safe distance for explosives is 30 feet. The safe distance for refueling operations is 45 feet.

WARNING

Remove all objects under the aircraft. Failure to do so can result in damage to aircraft or injury to personnel.

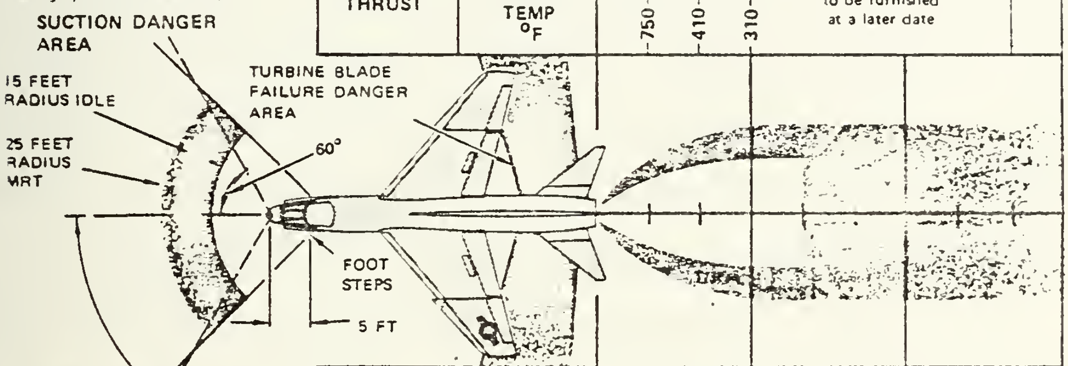
Stay clear of area within 100 feet directly behind the aircraft when the engine is operating at MRT.

LEGEND

- Military Thrust Danger Areas
- Idle Thrust Danger Areas
- Radiation Hazard

FEET 0 30 60 90

MILITARY THRUST	VELOCITY KNOTS	1185	711	533	Additional information to be furnished at a later date
	TEMP °F	750	410	310	

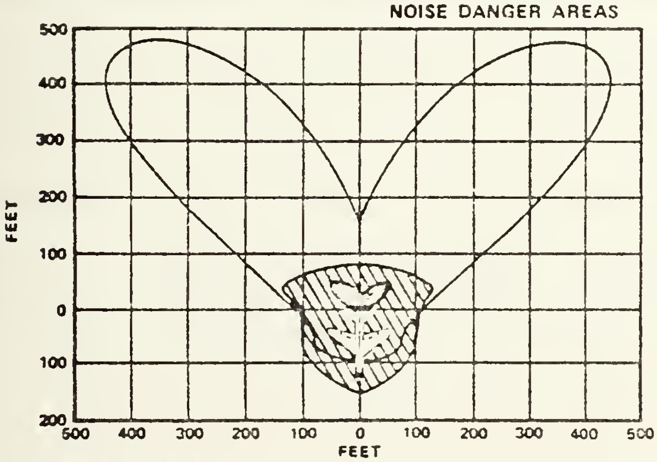


IDLE THRUST	TEMP °F	350	260	180
	VELOCITY KNOTS	267	178	118

CAUTION

Make certain that suction danger area is clear of debris. Jet blast zone varies according to prevailing wind.

NOISE DANGER AREAS



LEGEND

- Military Thrust — 140 DB and above. Dangerous even with ear protection. Limit exposure to absolute minimum.
- Idle Thrust — 95 DB and above. Ear Protection required.
- Military Thrust — 120 DB to 140 DB. Ear protection required. Use caution in exposure time.
- Military Thrust — 90 to 120 DB. Ear protection required (Limits of chart)

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Figure 1–74

APPENDIX B: JET EXHAUST TEST PLAN ON COMPOSITES

A. INTRODUCTION

This test series is to determine what conditions of jet exhuast impingement are required to cause structural damage to graphite-epoxy composites. Aircraft operational environment aboard aircraft carriers results in intermittent exposure of one aircraft to the jet exhaust of other aircraft. For conventional metal structured aircraft, this has not presented any serious excessive heat problems. Metal structures, being good heat conductors, are difficult to locally heat to high temperatures. Annealling of aircraft metals requires temperatures in excess of 600°F. Graphite-epoxy composite structures of new aircraft (F-18, AV-8B) are insulators, thus easily subject to local heating. They degrade at temperatures as low as 400°F. (Most military aircraft paints show no temperature discolorations until heated to above 500°F).

Various specimen parameters and engine test conditions will affect the response of the composite to jet blast exposure. Thickness and type of paint are the main specimen-controlling parameters. Engine power setting, distance between the engine and specimen, time duration of exposure, and specimen angle in the exhaust flow are the chief controlling test conditions.

B. TEST OUTLINE

Test No.	Specimen Thickness (inches)	Paint Color	Engine* Power (% Mil)	Separation* Distance (feet)	Angle of Attack (deg)	Expo-sure** Time (sec)
		SINGLE PLATE TEST CONDITIONS				
1	1/8	White	80	10	0	2
2	1/8	White	80	10	0	10
3	1/8	White	80	10	0	20
4	1/8	White	80	10	0	60
5	1/8	Dk Gray	80	10	0	2
6	1/8	Dk Gray	80	10	0	10
7	1/8	Dk Gray	80	10	0	20
8	1/4	White	80	10	0	2
9	1/4	White	80	10	0	10
10	1/2	White	80	10	0	2
11	1/2	White	80	10	0	10
		MULTIPLE PLATE TEST CONDITIONS				
12	1/8	White	80	10	0	10
	1/8	Lt Gray				
	1/8	Dk Gray				
13	1/4	White	80	10	0	10
	1/4	Lt Gray				
	1/4	Dk Gray				
14	1/2	White	80	10	0	10
	1/2	Lt Gray				
	1/2	Dk Gray				
15	1/2	White	80	10	0	2x5***
	1/2	Lt Gray				
	1/2	Dk Gray				
16	1/2	White	80	10	0	2x10***
	1/2	Lt Gray				
	1/2	Dk Gray				

*Engine power settings and separation distances subject to change based on final results of current analysis and results of first series of tests.

Test No.	Specimen Thickness (inches)	Paint Color	Engine* Power (% Mil)	Separation* Distance (feet)	Angle of Attack (deg)	Exposure** Time (sec)
17	1/8 1/4 1/2	White Lt Gray Dk Gray	80	10	0	Time to steady-state temperature
18	1/8 1/8 1/8	White Lt Gray Dk Gray	Idle	5	0	Time to steady-state temperature
19	1/4 1/4 1/4	White Lt Gray Dk Gray	Idle	5	0	Time to steady-state temperature
20	1/2 1/2 1/2	White Lt Gray Dk Gray	Idel	5	0	Time to steady-state temperature
21	1/8 1/8 1/8	White Lt Gray Dk Gray	80	20	0	15
22	1/4 1/4 1/4	White Lt Gray Dk Gray	80	20	0	15
23	1/2 1/2 1/2	White Lt Gray Dk Gray	80	20	0	15
24	1/8 1/4 1/4	Lt Gray Lt Gray lt Gray	80	20	0	Time to steady-state temperature
25	1/8 1/8 1/8	White Lt Gray Dk Gray	80	30	0	30

**Exposure times are approximate - thermocouple indicated temperatures will tend to control the duration fo some of these tests.

Test No.	Specimen Thickness (inches)	Paint Color	Engine* Power (% Mil)	Separation* Distance (feet)	Angle of Attack (deg)	Exposure ** Time (Sec)
26	1/4	White	80	30	0	30
	1/4	Lt Gray				
	1/4	Dk Gray				
27	1/2	White	80	30	0	30
	1/2	Lt Gray				
	1/2	Dk Gray				
28	1/8	Dk Gray	80	30	0	Time to steady-state temperature
	1/4	Dk Gray				
	1/2	Dk Gray				
29	1/8	Lt Gray	80	20	90	Time to steady-state temperature
	1/4	Lt Gray				
	1/2	Lt Gray				

The F-111 aircraft with TF-30 engines will be used to supply jet engine exhaust gases for this test series. The F-18 wing bos section will be used to mount the graphite-epoxy test specimens. The wing box is to be fitted with its monolithic aluminum lower wing skin. The upper aluminum skin will be modified to include cut out flush mounts for three test specimens. The wing box is to be fitted with existing steel leading and trailing edges. The assembly will mount on a test stand which will include a water-cooled, remote-operated blast deflector to protect the upper wing skin between jet exhaust exposures.

*** Cycles exposure with engine to idle and blast deflector protection between exposures while specimen cools to ambient temperature before next exposure.

A typical test will consist of connecting the thermocouple leads to the test specimen, then mounting it in the wing box. The jet blast deflector will be lowered over the wing and, following instrument checks, the aircraft engine will be started and idled. The engine will be advanced to the desired power setting and the test started by raising the jet blast deflector. Real-time thermocouple monitoring will indicate when to stop the test.

The first test of each working day will require about 1 hour set-up time to turn on and warm up all electronic equipment and to start up TF-30 engines. Thereafter, each test will take about 30 minutes for turn-around. It is estimated that 10 tests per day can be expected. Including cleanup, this test series will require 5 to 6 range days. It is recommended that the tests be conducted at the C-3 pad for ease in use of the F-111 aircraft engine testbed.

Lt. John Hampey of the Naval Postgraduate School will participate as an observer on at least one test day.

C. COORDINATION

Code 3383 individual responsibilities:

J. S. Fontenot - Project Manager - Alternate point of contact with Code 3383.

L. F. DeSandre - Test Engineer. Will provide engineering support. Will assist with instrumentation. Will receive all test data and notes at end of test. Main point of contact with Code 3383.

D. TEST SITE SET-UP

The F-111 will be tied to the test pad using existing holdback points. The F-18 wingbox will be positioned behind the aircraft such that the core exhaust centerline will impinge the wing leading edge directly in line with the test specimen. A television camera will be required to monitor real-time response of the test specimen.

E. HARDWARE LIST

1. Water hose with spray nozzle to be used on composite specimens if they ignite during any of the tests.

2. Start and safety support equipment for the F-111 aircraft.

3. F-18 wingbox section configured as follows:

a. Monolothic aluminum lower wing skin installed.

b. Upper aluminum wing skin to be modified such that (1) it will have three cutouts each 5 in. x 5 in. and (2) 1/2 inch wide by 1/2 inch deep borders around each cutout per Figure (1) with four holes drilled and tapped at each corner.

c. Hardwire type K thermocouple connectors (15 each) and allow sufficient slack in the leads such that all of the connectors will reach each of the three specimen mounting locations.

d. Install three air temperature indicating thermocouples in the upper wing skin per Figure (2). Make certain that thermocouple connectors are compatible with connectors installed in (c) and that there is sufficient slack to allow

connecting them through the specimen mounting cutouts after the wing skin has been installed on the wingbox.

e. Thermocouple leads to pass through the lower wing skin inside a protective steel pipe which is secured to the wing mounting stand.

f. Frabricate two aluminum panels which will duplicate test specimens. (See Figure 3). These panels will be used as closures for all of the wing skin cutouts not having a composite panel installed for a given test.

g. Shin rings, 3 each, at thicknesses of 3/8 inch and 1/4 inch per Figure 4.

4. Water-cooled jet blast deflector per Figure 5.

5. F-111 aircraft with provisions for remote control of the engines. Only one engine will probably be required for this test series; however, provisions for simultaneous operation of both engines will be desirable.

6. Jet fuel of type and quantity sufficient for this test series.

7. Protective clothing: see O.P. 3184.b, dtd 25 June 1979.

F. INSTRUMENTATION

Temperature recording and color television coverage of this test series will be required.

1. Temperature - any single test will require a minimum of 15 thermocouple channel recording (chromel-alumel, type K).

At least four of these channels must be displayed in real-time at the control center.

2. Television - at least one camera will be required. It should be mounted such that it gives a good close-up top view of each of the six specimen mounting locations. It must be located so that it is safely away from the jet exhaust and not have the jet blast deflector, when in the raised position, obscure view of the test specimens.

3. A hot wire anemometer may be used to measure jet exhaust velocities during some or all of these tests. The instrument will be furnished by Code 3383.

4. TF-30 engine monitoring equipment must include as a minimum a high accuracy digital percent power indicator and a real-time engine exhaust temperature (EGT) indicator. Steady state readings of these parameters will be required for each test.

5. Camera - still photographs will be required to document all testing. This service will be provided by Code 3383.

6. Adhesive - coated microscope slides shall be located at selected positions from the wing box and be periodically checked for excessive fiber contamination.

G. TEST PERSONNEL

Code	Title	Responsibilities
3383	Project Manager	On site during all tests. Monitor data during test. Insure that all test conditions are met. Decide when each test is complete.

Code	Title	Responsibilities
		Receive all test data and specimens at end of test. Supply painted and thermocoupled test specimens. Responsible for adequacy and completeness of testing.
3384	Safety Engineer	
3384	Mechanical Technician (2)	Preparation and set-up of wing box.
3384	Electronic Technician	Installation, hookup, and checkout of all thermocouples, connectors, and recorders except thermocouples in test specimen. Set up TV camera.
3384	Electronic Engineer	Recording all thermocouple data and television during each test.
3384	Propulsion Technician (2)	TF-30 startup, operation, and control. Record engine power and EGT during each test.

H. PRETEST READINESS EVALUATION

Initiation of any one of the tests in this series can occur when the following requirements have been met:

1. Test Specimens - installed in wingbos with all thermocouple connections made and continuity and polarity checked.

2. Wingbox - correctly positioned at desired distance from the engine exhaust nozzle, at the correct height and centered on the test specimen.

3. Jet Blast Deflector - remote operation verified, cooling water flow turned on, position set to LOW.

4. F-111 - tie-downs secured and checked. Engine started and remote throttle and engine monitoring instruments checked.

5. Television - camera installed, focused on the target specimens and recorder ready.

I. TEST PROCEDURE

1. Photograph and weigh each test specimen(s) (Code 3383).

2. Connect wingbox thermocouple leads to specimen(s) thermocouples.

3. Mount test specimen(s) in wingbox.

4. Position jet blast deflector and turn on water cooling.

5. Start TF-30 engine and warmup at IDLE power.

6. Accelerate engine to desired power setting.

7. For a given test condition extending over several days, make final engine power adjustments to maintain a fixed EGT.

8. Raise jet blast deflector.

9. Record test start time.

10. Record all engine parameters.

11. Monitor and record specimen thermocouple readings. When they reach predetermined temperature or predetermined time has elapsed, return engine power to IDLE.

12. Lower the jet blast deflector.

13. Continue to record specimen temperatures until they indicate ambient temperature.

NOTE: If composite specimen is burning at end of test, extinquish with water. Avoid breathing of smoke from such specimens.

14. Shut down engine.

15. Shut off recorders.

16. Photograph test specimen(s).

17. Unbolt and remove test specimens, taking care not to further damage heat-exposed face.

18. Place specimen in zip-lock bag with a card identifying the specimen and the conditions it was test at.

19. Weight specimen (Code 3383).

20. Store specimen for Project Engineer.

NOTE: All on-site personnel handling test specimens after exposure to jet blast shall wear protective clothing per O.P. 3184-8.b, dtd 25 June 79.

J. GENERAL POST-TEST SERIES REQUIREMENTS

General cleanup will include a thorough external water washdown of the wingbox, mounting stand, cabling, and desk around the test site to remove any residual carbon fibers from the area. The Code 3384 Branch Head will decide if any additional test site cleanup is required due to the possible presence of carbon fibers.

APPENDIX C

ASTM: 02344-72 APPARENT HORIZONTAL SHEAR STRENGTH OF REINFORCED PLASTIC BY SHORT BEAM METHOD*

A. SCOPE

This method covers the determination of the apparent horizontal shear strength of parallel fiber reinforced plastics. The specimen is a short beam in the form of segments cut from a ring-type specimen or a short beam cut from a flat laminate up to 6.4 mm (0.25 in.) in thickness. The method is applicable to all types of parallel fiber reinforced samples.

B. APPLICABLE DOCUMENTS

2.1 ASTM Standards:

D618, Conditioning Plastics and Electrical Insulating Materials for Testing**

D2991, Recommended Practice for Testing Stress-Relaxation of Plastics**

E4, Verification of Testing Machines***

E18, Tests for Rockwell Harnes and Rockwell Superficial Hardness of Metallic Materials****

*This method is under the jurisdiction of ASTM Committee D-30 on High Modulus Fibers and Their Composites. Current edition approved April 10, 1972. Published June 1972.

**Annual Book of ASTM Standards, Part 35.

C. SUMMARY OF METHOD

The horizontal shear test specimen (Fig. 2) is center-loaded as shown in Figures 15, 16, and 17. The specimen ends rest on two supports that allow lateral motion, the load being applied by means of a loading nose directly centered on the midpoint of the test specimen.

D. SIGNIFICANCE

Shear strength determined by this method is useful for quality control and specification purposes. It is also applicable for research and development programs concerned with interply strength. The apparent shear strength obtained in this method can not be used as a design criteria, but can be utilized for comparative testing of composite materials, if all failures are in horizontal shear.

The method is not limited to specimens with the sizes shown but is limited to specified span length-to-depth ratios. This ratio is recommended to be 5 when the specimen is reinforced with filaments having a Young's modulus of less than 100 x 10^9 Pa (14.5 x 10^6 psi) and 4 when the specimen is reinforced with filaments above 100 x 10^9 Pa (14.5 x 10^6 psi). See Table I for ratios for several typical reinforcements.

***Annual Book of ASTM Standards, Part 10, 14, 32, 35, and 41.

****Annual Book of ASTM Standards, Part 10

NOTE: The test method is also applicable to thicker specimens, especially where plies are thick (for example, ply thickness of 1.3 mm (0.05 in.) are sometimes seen in cloth reinforcements; it is only necessary to scale the fixture in proportion to the thickness).

E. APPARATUS

Testing machine, properly calibrated, which can be operated at constant rate of crosshead motion, and in which the error in the load measuring system shall not exceed $\pm$ 1 percent. The load-indicating mechanism shall be essentially free of inertia lag at the crosshead rate used. Inertia lag may not exceed 1 percent of the measured load. The accuracy of the testing machine shall be verified in accordance with Method E4.

Loading nose and supprts, as show in Figures 15 and 16. The loading nose shall be a 6.35-mm (0.250 in.) diameter dowel pin with a hardness of 60 to 62 HRC, as specified in Methods E18, and shall have a finely ground surface free of indentation and burrs with all sharp edges relieved.

Micrometers, suitable ball-type, reading to at least 0.025 mm (0.001 in.) for measuring the width, thickness, and length of the test specimen.

F. TEST SPECIMEN

The rings used in this test method shall be fabricated in accordance with Recommended Practice D 2291. The dimensions of the rings shall conform to the Type C specimens as described

in Recommended Practice @ 2291. Shear test specimens cut from the rings shall conform to the dimensions and notes specified in Figure 1.

NOTE: The flat specimens shall be molded by any suitable laminating means, such as press, bag, or autoclave molding.

The number of test specimens is optional. However, a minimum of ten specimens is required to obtain a satisfactory average for one ring or laminate.

G. CONDITIONING

Condition the test specimen and test in a room or enclosed space maintained at 23 ± 1°C (73.4 ± 1.8°F) and 50 ± 10 percent relative humidity in accordance with Procedure A of Methods D 618. Record any deviation from the above conditions.

If it is desired to test the effect of boiling water on the shear strength, place the specimens in boiling distilled water for a prescribed period of time; then remove and place in distilled water at 23 ± 1°C (73.4 ± 1.8°F) for a minimum of 15 min. Wipe the specimens dry and test at the standard conditions given above.

H. SPEED OF TESTING

Test the specimen at a rate of crosshead movement 1.3 mm (0.05 in.)/min.

I. PROCEDURE

Before conditioning or testing, measure the thickness and width of each specimen to the neast 0.025 mm (0.01 in.) at midpoint.

Place the test specimen in the test fixture as shown in Figures 15 or 16. Align the specimen so that it midpoint is centered and its long axis is perpendicular to the cylindrical axis or under the loading nose. Push the side supports into the span previously determined (depending on the modulus of the material being tested). Suggested span-to-depth ratios are given in Table I.

Apply the load to the specimen at the specified crosshead rate. Record the load to break specimen (maximum load on load-indicating mechanism). Often when testing laminates that are made with the high modulus fibers, specimens do not always fail in shear, especially when the incorrect span-to-depth ratio is chosen. It is therefore very important to record the type of break that occurs (shear or tensile). Also record the position of the shear plane (for example, left, right, center, or complete delamination across specimen.

J. RETESTS

Values for properties at break shall not be calculated for any specimen that breaks at some obvious, fortuitous flaw, unless such flaws constitute a variable being studied. Retests shall be made for any specimen on which values are not calculated. If a specimen in the shear test failed in a manner other than horizontal shear, the value shall be discarded and retest shall be made.

K. CALCULATIONS

Standard deviation - calculate the standard deviation (estimated) as follows and report to two significant figures:

$$s = (\sqrt{(\Sigma X^2 - n(\bar{X}))^2/(n-1)}$$

where

s = estimated standar deviation,

X = value of a single observation,

n = number of observations, and

$\bar{X}$ = arithmetic means of the set observations

TABLE VI

RECOMMENDED RATIO OF THICKNESS TO SPAN LENGTH AND TO SPECIMEN LENGTH

	SPAN/ THICKNESS	LENGTH/ THICKNESS
Woven cloth reinforcement	5	7
Continuous glass filaments	5	7
Silica fibers (continuous)	4	6
Graphite yarn	4	6
Carbon yarn	5	7
Boron filaments	4	6
Steel wire	5	7

APPENDIX D: RAW DATA FROM SHORT BEAM SHEAR TESTS

SHORT BEAM SHEAR TEST

DATE: 27 February 1981 TEST ASTM-D2344-72

MACHINE: INSTRON 014598 TESTED BY: J. M. Hampey

PAINT THICKNESS .005"

TEST 1A NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN. (IN)	BREAKING LOAD (lbf)
1A	.139	.850	.257	.5	410
1B	.141	.869	.269	.5	487
1C	.138	.869	.222	.5	392
1D	.138	.876	.232	.5	441
1E	.138	.878	.213	.5	400
1F	.138	.885	.220	.5	405
1G	.138	.865	.257	.5	455
1H	.138	.864	.214	.5	385

SHORT BEAM SHEAR TEST

DATE: 7 May 1981 TEST ASTM-D2344-72

MACHINE: INSTRON 014589 TESTED BY: J. M. Hampey

PAINT THICKNESS .033"

TEST 1B NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
1A	.135	.85	.255	.5	335
1B	.137	.8	.225	.5	450
1C	.136	.8	.230	.5	440
1D	.135	.85	.253	.5	410
1E	.136	.85	.253	.5	415
1F	.136	.85	.245	.5	380
1G	.133	.85	.278	.5	450
1H	.135	.85	.230	.5	380
1I	.136	.85	.237	.5	425
1J	.136	.85	.197	.5	315
1K	.136	.85	.234	.5	385
1L	.136	.85	.243	.5	415
1M	.135	.85	.234	.5	392
1N	.136	.85	.237	.5	370
1O	.135	.85	.221	.5	385
1P	.136	.85	.229	.5	380
1Q	.134	.85	.245	.5	410

SHORT BEAM SHEAR TEST

DATE: 26 March 1981

TEST ASTM-D2344-72

MACHINE: INSTRON 014598

TESTED BY: J. M. Hampey

PAINT THICKNESS .003

TEST NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
5A	.138	.81	.269	.5	515
5B	.136	.81	.258	.5	482
5C	.134	.82	.233	.5	350
5D	.134	.81	.234	.5	445
5E	.137	.82	.265	.5	535
5F	.135	.81	.267	.5	507
5G	.136	.81	.270	.5	507
5H	.137	.81	.265	.5	477
5I	.137	.81	.264	.5	515
5J	.137	.81	.270	.5	490
5K	.137	.81	.285	.5	483
5L	.135	.81	.265	.5	455
5M	.135	.81	.319	.5	577

SHORT BEAM SHEAR TEST

DATE: 26 March 1981 TEST ASTM-D2344-72

MACHINE: INSTRON 014589 TESTED BY: J. M. Hampey

PAINT THICKNESS .003

TEST NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
6A	.138	.76	.157	.5	290
6B	.138	.76	.241	.5	472
6C	.136	.76	.248	.5	460
6D	.139	.76	.250	.5	462
6E	.139	.76	.261	.5	488
6F	.139	.76	.267	.5	490
6G	.139	.76	.266	.5	495
6H	.139	.76	.264	.5	475
6I	.136	.76	.261	.5	475
6J	.136	.76	.273	.5	445
6K	.138	.76	.268	.5	505
6L	.136	.76	.268	.5	490
6M	.138	.76	.269	.5	483

SHORT BEAM SHEAR TEST

DATE: 26 March 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
7A	.137	.86	.211	.5	339
7B	.137	.86	.219	.5	300
7C	.136	.86	.241	.5	410
7D	.133	.86	.252	.5	476
7E	.135	.86	.243	.5	473
7F	.136	.86	2.43	.5	445
7G	.137	.86	.242	.5	386
7H	.132	.86	.256	.5	404
7I	.137	.86	.262	.5	462
7J	.134	.86	.261	.5	451
7K	.136	.86	.266	.5	450
7L	.136	.86	.264	.5	464
7M	.136	.86	.263	.5	415
7N	.136	.86	.266	.5	495

SHORT BEAM SHEAR TEST

DATE: 26 March 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
8A	.137	.83	.251	.5	368
8B	.139	.83	.249	.5	525
8C	.139	.83	.263	.5	550
8D	.138	.83	.261	.5	593
8E	.136	.83	.249	.5	428
8F	.137	.83	.260	.5	395
8G	.137	.83	.266	.5	455
8H	.136	.83	.257	.5	410
8I	.137	.83	.265	.5	455
8J	.139	.83	.266	.5	530
8K	.138	.83	.269	.5	530
8L	.137	.83	.260	.5	465
8M	.138	.83	.262	.5	543
8N	.136	.83	.292	.5	487

SHORT BEAM SHEAR TEST

DATE: 26 March 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
9A	.136	.87	.251	.5	430
9B	.134	.82	.255	.5	420
9C	.134	.82	.271	.5	498
9D	.135	.87	.260	.5	437
9E	.134	.87	.238	.5	387
9F	.134	.87	.248	.5	497
9G	.134	.87	.233	.5	473
9H	.135	.82	.268	.5	475
9I	.134	.87	.255	.5	432
9J	.134	.82	.228	.5	394
9K	.136	.82	.235	.5	412
9L	.136	.82	.243	.5	425
9M	.135	.87	.260	.5	480
9N	.133	.87	.244	.5	482

SHORT BEAM SHEAR TEST

DATE: 27 March 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN(IN)	BREAKING LOAD(1bf)
10A	.139	.83	.261	.5	565
10B	.139	.83	.252	.5	445
10C	.139	.83	.259	.5	502
10D	.139	.83	.260	.5	532
10E	.139	.83	.263	.5	542
10F	.139	.83	.253	.5	467
10G	.139	.83	.273	.5	615
10H	.138	.83	.261	.5	560
10I	.139	.83	.257	.5	490
10J	.143	.83	.245	.5	548
10K	.138	.83	.272	.5	543
10L	.138	.83	.241	.5	535
10M	.138	.83	.248	.5	500
10N	.139	.83	.264	.5	600
10O	.138	.83	.271	.5	528

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
11A	.138	.82	.258	.5	535
11B	.139	.82	.267	.5	457
11C	.138	.82	.263	.5	437
11D	.138	.82	.260	.5	440
11E	.139	.82	.271	.5	540
11F	.138	.82	.251	.5	540
11G	.140	.82	.254	.5	465
11H	.138	.82	.263	.5	485
11I	.138	.82	.251	.5	480
11J	.139	.82	.259	.5	450
11K	.138	.82	.252	.5	495
11L	.140	.82	.249	.5	455
11M	.140	.82	.260	.5	490
11N	.140	.82	.259	.5	550
11O	.138	.82	.241	.5	420
11P	.138	.82	.248	.5	465

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
12A	.134	.82	.275	.5	470
12B	.134	.82	.243	.5	370
12C	.134	.82	.236	.5	475
12D	.134	.82	.250	.5	445
12E	.134	.82	.257	.5	455
12F	.134	.82	.248	.5	450
12G	.134	.82	.249	.5	445
12H	.135	.82	.247	.5	450
12I	.133	.82	.244	.5	440
12J	.134	.82	.238	.5	432
12K	.133	.82	.253	.5	462
12L	.134	.82	.266	.5	500
12M	.135	.82	.236	.5	420
12N	.134	.82	.250	.5	428
12O	.134	.82	.250	.5	440
12P	.133	.82	.250	.5	463

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J.M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
13A	.139	.82	.257	.5	595
13B	.137	.82	.252	.5	600
13C	.139	.82	.235	.5	435
13D	.139	.82	.241	.5	470
13E	.138	.82	.255	.5	560
13F	.139	.82	.247	.5	470
13G	.139	.82	.215	.5	435
13H	.139	.82	.238	.5	500
13I	.137	.82	.242	.5	465
13J	.132	.82	.231	.5	495
13K	.139	.82	.254	.5	535
13L	.139	.82	.248	.5	465
13M	.134	.82	.241	.5	420
13N	.137	.82	.249	.5	470
13O	.136	.82	.244	.5	560
13P	.139	.82	.238	.5	505

SHORT BEAM SHEAR TEST

DATE: 7 May 1981 — TEST ASTM D2344-72

MACHINE: INSTRON — TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
1A	.274	1.5	.255	1.0	795
1B	.274	1.5	.260	1.0	935
1C	.274	1.5	.248	1.0	960
1D	.273	1.5	.253	1.0	900
1E	.274	1.5	.260	1.0	955
1F	.275	1.5	.270	1.0	1060
1G	.274	1.5	.261	1.0	840
1H	.275	1.5	.276	1.0	1100
1I	.275	1.5	.273	1.0	950
1J	.274	1.5	.262	1.0	945
1K	.274	1.5	.269	1.0	945
1L	.275	1.5	.257	1.0	970
1M	.274	1.5	.266	1.0	870
1N	.274	1.5	.270	1.0	1080
1O	.274	1.5	.258	1.0	890
1P	.275	1.5	.258	1.0	1035

SHORT BEAM SHEAR TEST

DATE: 9 March 1981 — TEST ASTM D2344-72

MACHINE: INSTRON — TESTED BY: J. M. Hampey

PAINT THICKNESS .005"

TEST NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
2A	.278	1.5	.291	1	758
2B	.275	1.65	.268	1.	956
2C	.275	1.65	.242	1	838
2D	.278	1.65	.242	1	808
2E	.278	1.65	.229	1	728
2F	.278	1.65	.272	1	912
2G	.275	1.65	.279	1	808
2H	.276	1.65	.323	1	998

SHORT BEAM SHEAR TEST

DATE: 9 March 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NUMBER	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN (IN)	BREAKING LOAD(lbf)
3A	.272	1.5	.288	1	840
3B	.279	1.65	.347	1	OFF SCALE
3C	.272	1.65	.245	1	965
3D	.272	1.65	.231	1	870
3E(NOTCHED)	.272	1.65	.257	1	450
3F	.272	1.65	.258	1	922
3G	.272	1.65	.249	1	820
3H	.273	1.5	.359	1	1160

SHORT BEAM SHEAR TEST

DATE: 9 March 1981 | TEST ASTM D2344-72

MACHINE: INSTRON | TESTED BY J.M. Hampey

PAINT THICKNESS .003"

TEST NUMBER	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
4A	.250	1.38	.258	1	140
4B	.250	1.44	.330	1	185
4C	.255	1.38	.307	1	210
4D	.248	1.50	.232	1	115
4E	.253	1.50	.230	1	135
4F	.255	1.50	.226	1	100
4G	.264	1.50	.330	1	165
4H	.262	1.50	.237	1	115

SHORT BEAM SHEAR TEST

DATE: 27 March 1981 TEST ASTM D-2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
5A	.273	1.5	.255	1	1040
5B	.273	1.5	.279	1	1050
5C	.272	1.5	.265	1	1012
5D	.272	1.5	.244	1	800
5E	.273	1.5	.263	1	980
5F	.273	1.5	.268	1	1040
5G	.272	1.5	.257	1	1070
5H	.272	1.5	.278	1	1065
5I	.272	1.5	.280	1	1085
5J	.272	1.5	.275	1	1038
5K	.272	1.5	1250	1	1000
5L	.271	1.5	.292	1	1080
5M	.271	1.5	.262	1	940
5N	.272	1.5	.310	1	1220
5O	.272	1.5	.245	1	840

SHORT BEAM SHEAR TEST

DATE: 1 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
6A	.273	1.5	.251	1.0	900
6B	.273	1.5	.262	1.0	1035
6C	.270	1.5	.256	1.0	840
6D	.271	1.5	.253	1.0	1050
6E	.274	1.5	.253	1.0	790
6F	.273	1.5	.241	1.0	920
6G	.273	1.5	.251	1.0	1020
6H	.275	1.5	.253	1.0	880
6I	.273	1.5	.264	1.0	970
6J	.269	1.5	.248	1.0	800
6K	.269	1.5	.264	1.0	950
6L	.271	1.5	.259	1.0	1050
6M	.273	1.5	.241	1.0	950
6N	.269	1.5	.241	1.0	900
6O	.272	1.5	.246	1.0	1050
6P	.270	1.5	.200	1.0	700

SHORT BEAM SHEAR TEST

DATE: 27 March 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
7A	.271	1.5	.256	1	870
7B	.271	1.5	.257	1	867
7C	.274	1.5	.262	1	1050
7D	.273	1.5	.272	1	1000
7E	.268	1.5	.269	1	1060
7F	.265	1.5	.262	1	1080
7G	.266	1.5	1263	1	995
7H	.274	1.5	.275	1	1100
7I	.273	1.5	.256	1	1050
7J	.273	1.5	.295	1	1120
7K	.274	1.5	.232	1	810
7L	.272	1.5	.248	1	980
7M	.271	1.5	.261	1	865
7N	.271	1.5	.357	1	1335
7O	.262	1.5	.361	1	1338

SHORT BEAM SHEAR TEST

DATE: 1 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
8A	.272	1.5	.252	1.0	940
8B	.272	1.5	.258	1.0	900
8C	.274	1.5	.261	1.0	1085
8D	.272	1.5	.260	1.0	935
8E	.270	1.5	.257	1.0	1045
8F	.263	1.5	.254	1.0	945
8G	.264	1.5	.250	1.0	970
8H	.272	1.5	.252	1.0	900
8I	.273	1.5	.265	1.0	1100
8J	.274	1.5	.258	1.0	1070
8K	.274	1.5	.266	1.0	1130
8L	.270	1.5	.262	1.0	1070
8M	.259	1.5	.276	1.0	990
8N	.269	1.5	.262	1.0	980
8O	.266	1.5	.261	1.0	890
8P	.271	1.5	.240	1.0	1000

SHORT BEAM SHEAR TEST

DATE: 1 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
9A	.275	1.5	.255	1.0	1140
9B	.275	1.5	.263	1.0	930
9C	.275	1.5	.262	1.0	990
9D	.274	1.5	.246	1.0	975
9E	.274	1.5	.290	1.0	1045
9F	.274	1.5	.245	1.0	890
9G	.274	1.5	.269	1.0	1020
9H	.274	1.5	.265	1.0	990
9I	.275	1.5	.237	1.0	890
9J	.273	1.5	.262	1.0	900
9K	.274	1.5	.270	1.0	1035
9L	.273	1.5	.282	1.0	1050
9M	.274	1.5	.256	1.0	990
9N	.273	1.5	.262	1.0	920
9O	.273	1.5	.249	1.0	900

SHORT BEAM SHEAR TEST

DATE: 1 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
10A	.265	1.5	.260	1.0	940
10B	.263	1.5	.248	1.0	840
10C	.264	1.5	.263	1.0	1005
10D	.263	1.5	.269	1.0	915
10E	.265	1.5	.283	1.0	965
10F	.265	1.5	.250	1.0	830
10G	.265	1.5	.247	1.0	890
10H	.266	1.5	.268	1.0	1025
10I	.263	1.5	.242	1.0	965
10J	.264	1.5	.246	1.0	875
10K	.263	1.5	.256	1.0	890
10L	.264	1.5	.256	1.0	785
10M	.263	1.5	.242	1.0	945
10N	.263	1.5	.257	1.0	780
10O	.264	1.5	260	1.0	840

SHORT BEAM SHEAR TEST					
DATE: 14 April 1981			TEST ASTM D2344-72		
MACHINE: INSTRON			TESTED BY: J. M. Hampey		
PAINT THICKNESS .003"					
TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
11A	.276	1.5	.233	1.0	905
11B	.275	1.5	.246	1.0	970
11C	.276	1.5	.241	1.0	830
11D	.276	1.5	.249	1.0	1000
11E	.273	1.5	.252	1.0	1000
11F	.275	1.5	.252	1.0	1010
11G	.274	1.5	.265	1.0	1030
11H	.273	1.5	.252	1.0	935
11I	.274	1.5	.275	1.0	1020
11J	.273	1.5	.230	1.0	975
11K	.276	1.5	.258	1.0	900
11L	.274	1.5	.254	1.0	940
11M	.276	1.5	.251	1.0	950
11N	.275	1.5	.251	1.0	860
11O	.275	1.5	.246	1.0	930
11P	.276	1.5	.257	1.0	1000
11Q	.273	1.5	.252	1.0	1120

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .005"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SAPN (IN)	BREAKING LOAD (lbf)
12A	.270	1.5	.264	1.0	740
12B	.270	1.5	.252	1.0	825
12C	.271	1.5	.252	1.0	990
12D	.272	1.5	.258	1.0	935
12E	.269	1.5	.248	1.0	680
12F	.274	1.5	.258	1.0	1040
12G	.272	1.5	.267	1.0	1080
12H	.273	1.5	.248	1.0	990
12I	.272	1.5	.263	1.0	1045
12J	.270	1.5	.246	1.0	850
12K	.269	1.5	.226	1.0	670
12L	.273	1.5	.248	1.0	960
12M	.270	1.5	.249	1.0	895
12N	.273	1.5	.253	1.0	965
12O	.272	1.5	.262	1.0	1010
12P	.273	1.5	.254	1.0	1005

SHORT BEAM SHEAR TEST					
DATE: 14 April 1981			TEST ASTM D2344-72		
MACHINE: INSTRON			TESTED BY: J. M. Hampey		
PAINT THICKNESS .003"					
TEST NO.	THICKNESS SPEC. (IN)	LENGTH M SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
13A	.274	1.5	.247	1.0	1095
13B	.274	1.5	.247	1.0	1050
13C	.274	1.5	.241	1.0	795
13D	.274	1.5	.232	1.0	845
13E	.272	1.5	.253	1.0	1050
13F	.271	1.5	.263	1.0	885
13G	.273	1.5	.246	1.0	960
13H	.275	1.5	.255	1.0	1050
13I	.271	1.5	.246	1.0	780
13J	.272	1.5	.235	1.0	830
13K	.272	1.5	.271	1.0	1015
13L	.273	1.5	.260	1.0	965
13M	.273	1.5	.239	1.0	1060
13N	.272	1.5	.259	1.0	1000
13O	.273	1.5	.258	1.0	940
13P	.272	1.5	.226	1.0	875

SHORT BEAM SHEAR TEST

DATE: 7 May 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NUMBER	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
1A	.557	3.0	.234	2.1	1620
1B	.556	3.0	.238	2.1	1540
1C	.555	3.0	.272	2.1	1780
1D	.554	3.0	.262	2.1	1710
1E	.557	3.0	.230	2.1	1560
1F	.549	3.0	.239	2.1	1410
1G	.555	3.0	.259	2.1	1690
1H	.554	3.0	.264	2.1	1700
1I	.555	3.0	.269	2.1	1760
1J	.554	3.0	.237	2.1	1580

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
5A	.539	3.1	.233	2.1	1650
5B	.538	3.1	.269	2.1	1910
5C	.539	3.1	.250	2.1	1660
5D	.536	3.1	.247	2.1	1640
5E	.535	3.1	.262	2.1	1730
5F	.539	3.1	.250	2.1	1715
5G	.539	3.1	.240	2.1	1530
5H	.538	3.1	.236	2.1	1560
5I	.538	3.1	.266	2.1	1650
5J	.536	3.1	.243	2.1	1570
5K	.539	3.1	.302	2.1	2100

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J.M. Hampey

NO PAINT AS LAYER SEPATATED

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
6A	.545	2.9	.260	2.1	1130
6B	.540	2.9	.258	2.1	680
6C	.543	2.9	.256	2.1	350
6D	.545	.29	.305	2.1	1230
6E	.545	2.9	.260	2.1	1170
6F	.544	2.9	.264	2.1	1200
6G	.545	.29	.272	2.1	1180
6H	.540	2.9	.254	2.1	930
6I	.546	2.9	.254	2.1	990
6J	.541	2.9	.274	2.1	1090
6K	.546	2.9	.261	2.1	1150
6L	.542	2.9	.274	2.1	600
6M	.546	2.9	.256	2.1	1050
6N	.537	2.9	.360	2.1	450

SHORT BEAM SHEAR TEST

DATE: 14 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
7A	.554	3.1	.255	2.1	1920
7B	.554	3.1	.262	2.1	1925
7C	.554	3.1	.271	2.1	1880
7D	.552	3.1	.253	2.1	1690
7E	.553	3.1	.264	2.1	1760
7F	.556	3.1	.265	2.1	1880
7G	.554	3.1	.247	2.1	1670
7H	.554	3.1	.256	2.1	1880
7I	.553	3.1	.265	2.1	1790
7J	.554	3.1	.256	2.1	1875

SHORT BEAM SHEAR TEST

DATE: 18 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD(lbf)
8A	.556	3.25	.251	.1	1660
8B	.556	3.25	.266	2.1	1780
8C	.556	3.25	.245	2.1	1500
8D	.554	3.0	.246	2.1	1590
8E	.554	3.0	.256	2.1	1750
8F	.554	3.0	.268	2.1	1800
8G	.555	3.0	.248	2.1	1690
8H	.555	3.0	.251	2.1	1780
8I	.553	3.0	.262	2.1	1795
8J	.553	3.0	.225	2.1	1400

SHORT BEAM SHEAR TEST

DATE: 15 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
9A	.555	3.0	.259	2.1	1690
9B	.554	3.0	.264	2.1	1700
9C	.555	3.0	.269	2.1	1760
9D	.554	3.0	.237	2.1	1580
9E	.557	3.0	.234	2.1	1620
9F	.556	3.0	.238	2.1	1540
9G	.555	3.0	.272	2.1	1780
9H	.554	3.0	.262	2.1	1710
9I	.557	3.0	.230	2.1	1560
9J	.549	.275	.239	2.1	1410

SHORT BEAM SHEAR TEST

DATE: 15 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
10A	.538	2.9	.269	2.1	1770
10B	.539	2.9	.272	2.1	1850
10C	.538	2.9	.257	2.1	1770
10D	.537	2.9	.248	2.1	1580
10E	.537	2.9	.237	2.1	1540
10F	.540	2.9	.249	2.1	1430
10G	.539	2.9	.258	2.1	1740
10H	.539	2.9	.256	2.1	1800
10I	.538	2.9	.283	2.1	1830
10J	.539	2.9	.246	2.1	1720

SHORT BEAM SHEAR TEST

DATE: 15 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(NI)	WIDTH SPEC.(IN)	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
11A	.559	3.25	.280	2.1	1700
11B	.559	3.25	.236	2.1	1450
11C	.559	3.25	.255	2.1	1450
11D	.556	3.0	.258	2.1	1595
11E	.554	3.0	.255	2.1	1690
11F	.551	3.0	.249	2.1	1620
11G	.553	3.0	.265	2.1	1700
11H	.553	3.0	.251	2.1	1520
11I	.554	3.0	.249	2.o	1575
11J	.553	3.0	.252	2.1	1630

SHORT BEAM SHEAR TEST

DATE: 15 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .003"

TEST NO.	THICKNESS SPEC.(IN)	LENGTH SPEC.(IN)	WIDTH SPEC.(IN)N	LENGTH SPAN(IN)	BREAKING LOAD(lbf)
12A	.547	3.0	.261	2.1	1635
12B	.548	3.0	.245	2.1	1660
12C	.548	3.0	.263	2.1	1770
12D	.548	3.0	.259	2.1	1570
12E	.548	3.0	.251	2.1	1650
12F	.549	3.0	.251	2.1	1765
12G	.547	3.0	.245	2.1	1530
12H	.548	3.0	.247	2.1	1420
12I	.547	3.0	.253	2.1	1570
12J	.549	3.0	.255	2.1	1760

SHORT BEAM SHEAR TEST

DATE: 15 April 1981 TEST ASTM D2344-72

MACHINE: INSTRON TESTED BY: J. M. Hampey

PAINT THICKNESS .004"

TEST NO.	THICKNESS SPEC. (IN)	LENGTH SPEC. (IN)	WIDTH SPEC. (IN)	LENGTH SPAN (IN)	BREAKING LOAD (lbf)
13A	.548	3.0	.260	2.1	1650
13B	.547	3.0	.261	2.1	1570
13C	.548	310	.249	2.1	1560
13D	.547	3.0	.255	2.1	1540
13E	.547	3.0	.258	2.1	1560
13F	.550	3.0	.267	2.1	1750
13G	.548	3.0	.240	2.1	1510
13H	.549	3.0	.276	2.1	1720
13I	.547	3.0	.265	2.1	1730
13J	.549	3.0	.238	2.1	1530

SHEAR STRESS CALCULATIONS 1/2" SAMPLES					
SAMPLE LETTER	TEST 1	TEST 5	TEST 6	TEST 7	TEST 8
A	9370	9910	5980	10250	8970
B	8780	9950	3660	10000	9070
C	8890	9290	1890	9440	8300
D	8880	9340	5550	9120	8800
E	9180	9310	6190	9110	9300
F	8100	9600	6270	9620	9140
G	8860	8920	5970	9240	9260
H	8760	9270	5080	10030	9640
I	8890	8700	5350	9230	9340
J	9070	9090	5510	9970	8480
K		NOTCHED SPECIMEN	6050		
L			3030		
M			5630		
N			HOLE IN SPECIMEN		
O					
P					
Q					
R					
NUMBER SAMPLES	10	10	13	10	10
TOTAL	88800	93380	66180	96010	90310
MEAN	8880	9340	5090	9600	9030
STANDARD DEVIATION	330	400	1370	430	410

SHEAR STRESS CALCULATIONS 1/2" SAMPLES					
SAMPLE LETTER	TEST 9	TEST 10	TEST 11	TEST 12	TEST 13
A	8860	9220	8190	8640	8730
B	8760	9520	8290	9320	8290
C	8890	9520	7670	9260	8620
D	9070	8950	8380	8340	8330
E	9370	9130	9020	9050	8340
F	8780	8020	8900	9660	8990
G	8890	9440	8750	8610	8660
H	8880	9840	8260	7910	8560
I	9180	9060	8610	8560	9000
J	8100	9780	8820	9480	8830
K					
L					
M					
N					
O					
P					
Q					
R					
NUMBER SAMPLE	10	10	10	10	10
TOTAL	88800	92610	84890	88830	88800
MEAN	8880	9260	8490	8880	8880
STANDARD DEVIATION	330	530	410	560	330

SHEAR STRESS CALCULATION 1/4" SAMPLES					
SAMPLE LETTER	TEST 1	TEST 2	TEST 3	TEST 4	TEST 5
A	8630	POOR CUT	8130	1630	11330
B	9950	9910	OFF SCALE	1680	10450
C	10710	9620	9840	2010	10650
D	9880	9170	10500	2000	9140
E	10160	8730	POOR CUT	1740	10350
F	10820	9210	9960	1300	10780
G	8910	8040	9180	1420	11610
H	10990	8550	8970	1390	10680
I	9600				10800
J	9980				10520
K	9720				11150
L	10410				10350
M	9050				10040
N	11070				10970
O	9550				9560
P	11060				
Q					
R					
NUMBER SAMPLE	16	8	8	8	15
TOTAL	160500	63240	56600	13170	158390
MEAN	10030	9030	9430	1650	10560
STANDARD DEVIATION	780	890	840	270	640

SHEAR STRESS CALCULATIONS 1/4" SAMPLES					
SAMPLE LETTER	TEST 6	TEST 7	TEST 8	TEST 9	TEST 10
A	9960	9510	10400	12330	10350
B	10970	9440	9720	9750	9770
C	9220	11090	11500	10420	10980
D	11610	10210	10030	10970	9810
E	8640	11150	11420	9970	9760
F	10600	11800	10730	10050	9500
G	11290	10790	11150	10490	10310
H	9590	11070	9960	10340	10910
I	10210	11390	11530	10350	11500
J	9100	10550	11480	9540	10220
K	10150	9660	11760	10610	10030
L	11340	11020	11470	10340	8810
M	10950	9270	10510	10700	11260
N	10530	10460	10550	9750	8750
O	11900	10730	9720	10040	9280
P	9830		11660		
Q					
R					
NUMBER SAMPLE	16	15	16	15	15
TOTAL	158390	158160	173590	155670	151260
MEAN	10560	10540	10850	10380	10080
STANDARD DEVIATION	640	770	730	660	830

SHEAR STRESS CALCULATIONS 1/4" SAMPLES					
SAMPLE LETTER	TEST 11	TEST 12	TEST 13		
A	10670	7930	12270		
B	10870	9260	11760		
C	9460	11080	9130		
D	11030	10180	10080		
E	11020	7790	11570		
F	11050	11240	9420		
G	10760	11360	10840		
H	10310	11170	11350		
I	10260	11160	8870		
J	11780	9780	9850		
K	9580	8420	10440		
L	10240	10830	10310		
M	10400	10170	12320		
N	9450	10670	10760		
O	10420	10830	10120		
P	10690	10990	10790		
Q	12340				
R					
NUMBER SAMPLE	17	16			
TOTAL	180350	162880			
MEAN	10610	10180			
STANDARD DEVIATION	760	1210			

SHEAR STRESS CALCULATIONS 1/8" SAMPLES					
SAMPLE LETTER	TEST 1A	TEST 1B	TEST 5	TEST 6	TEST 7
A	8930	SAMPLE CUT WRONG	10640	10260	8990
B	9980	11190	10540	10880	7670
C	9960	10790	8600	10460	9590
D	10720	9210	10890	10190	10900
E	10590	9250	11300	10310	11060
F	10380	8750	10790	10120	10330
G	9980	9340	10590	10260	8930
H	10140	9390	10070	9920	9180
I		10110	10920	10260	9870
J		9020	10160	9190	9890
K		9280	9480	10470	9540
L		9630	9760	10310	9910
M		9520	10120	9980	8900
N		8800			10490
O		9900			
P		9360			
Q		9580			
R					
NUMBER SAMPLES	8	16	13	13	14
TOTAL	80687.2	153110	133850	132620	135240
MEAN	10090	9570	10300	10200	9660
STANDARD DEVIATION	550	660	720	390	900

SHEAR STRESS CALCULATION 1/8" SAMPLES					
SAMPLE LETTER	TEST 8	TEST 9	TEST 10	TEST 11	TEST 12
A	8200	9660	11940	11520	9780
B	11630	9340	9740	9440	8720
C	11530	10520	10690	9230	11520
D	12630	9550	11280	9400	10190
E	9690	9310	11360	10990	10140
F	8500	11470	10180	11950	10390
G	9570	11620	12420	10020	10230
H	9000	10070	11920	10240	10350
I	9610	9700	10510	10620	10400
J	10990	9890	11980	9580	10390
K	10940	9890	11090	10910	10530
L	10010	9860	12330	10000	10760
M	11510	10490	11200	10320	10110
N	9400	11400	12530	11620	9800
O			10820	9680	10080
P				10420	10680
Q					
R					
NUMBER SAMPLE	14	14	15	16	16
TOTAL	143230	142860	170010	165960	164090
MEAN	10230	10200	11330	10370	10260
STANDAR DEVIATION	1320	781	850	840	580

SHEAR STRESS CALCULATION 1/8" SAMPLES					
SAMPLE LETTER	TEST 13				
A	12770				
B	13330				
C	10210				
D	10750				
E	12200				
F	10490				
G	11160				
H	11580				
I	10750				
J	12460				
K	11620				
L	10340				
M	9980				
N	10560				
O	12940				
P	11700				
Q					
R					
NUMBER SAMPLES	16				
TOTAL	182850				
MEAN	11430				
STANDARD DEVIATION	1060				

APPENDIX E: GLOSSARY OF TERMS

Lamina - a flat (sometimes curved as in a shell) arrangement of undirectional fibers or woven fibers in a matrix.

Laminate - a stack of laminae with various orientations of principal material directions in the laminae.

Layup - the arranging of fibers in laminae and laminae in layers or laminates.

Curing - the drying, or polymerization, of the resinous matrix material to form a permanent bond between fibers and between laminae.

LIST OF REFERENCES

1. Composite Materials Volume 3 Engineering Applications Edited by Bryan R. Noton, Academic Press, 1974, pp. 121-159.

2. Glen and Springer, "Effects of Moisture and Temperature on the Tensile Strength of Composite Materials," Journal of Composite Materials, Volume 11, pp. 2 - 15, January 1977.

3. Whitney and Kin, "High Temperature Tensile Strength of Graphite/Epoxy Laminates Containing Circular Holes," Journal of Composite Materials, Vol. 10, pp. 319 - 324, October 1976.

4. Holman, P. P., Heat Transfer, McGraw-Hill, 1976.

5. Eckert, E. R. G. and Drake, R. M., Analysis of Heat and Mass Transfer, McGraw-Hill, 1972.

6. Wilson, D. W., "Behavior of Composite Bolted Joints at Elevated Temperature," NASA CP 159137, September 1979.

7. 30th Anniversay Technical Conference, 1975 Reinforced Plastics/Composites Institute, The Society of the Plastics Industry, Inc., Time-Temperature Moisture Studies on Graphite Fiber Reinforced Epoxy Composites, by Chung and Cragnola, Section 9-A, pp. 1 - 5.

8. 30th Anniversary Technical Conference, 1975 Reinforced Plastics/Composites Institute, The Society of the Plastics Industry, Inc., The Effects of Thermal Cycling on Advanced Composites Angle Ply Laminates, by Lebir, T., Daniel, I. M. and Chamis, C.C., Section 18-b, pp 1 - 6.

9. Composite Materials in Engineering Design Symposium, 6th Proceedings, High Temperature Strength Degradation of Composite During Aging in the Ambient Atmosphere, by Young, J. L., and Greever, W. L., pp. 695 - 716, May 11-12, 1972.

10. ASTM Special Technical Publication Number 636, Effect of Temperature on Fatigue Response of Surface-Notched [(0/+45/0)], Graphite/Epoxy Laminate, by Sendesky, G. P., Stalnaker, H. D., and Kleismit, R. A., pp. 123 - 140, 1977.

12. Sharma, A. V., "Effect of Temperature on Composite Sandwhich Structures Subjected to Low Velocity Impact," North Carolina Agricultural and Technical Papers, February 1980.

13. Jones, R. M. Mechanics of Composite Materials, McGraw-Hill, 1975.

14. Gruman Aerospace Corporation, NATOPS Flight Manual Navy Model F-14A Aircraft, July 1978.

15. NAVAIRENGCEN TEST DEPT. LETTER REPORT, UNCLASSIFIED NO. NAEC-TL232 to Commander, Naval Air Systems Command (AIR-551) Subject: F-18A Aircraft Contractor's JBD (Jet Blast Deflector) Demonstration Tests: Phase I, 9 November 1979.

16. NAVAIRENGCEN TEST DEPT. LETTER REPORT, UNCLASSIFIED NO: NAEC-TL233 to Commander, Naval Air Systems Command (Air-551) Subject: F-18A Aircraft Contractor's JBD (Jet Blast Deflector) Demonstration Tests: Phase II, 29 April 1980.

17. NAVAIRTESTFAC LETTER REPORT, UNCLASSIFIED NO: NATF-L176 To Commander, Naval Air Systems Command (Air-537) Subject: F-14A Aircraft Contractor's Jet Blast Deflector Demonstration Tests, 20 March 1974.

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5.	LT. John M. Hampey, USN 133 Chestnut Hill Lane West Reisterstown, Maryland 21136	3
6.	Mr. John Fontenot, Code 3383 Survivability and Lethality Division Systems Development Department Naval Weapons Center China Lake, California 93555	2

Thesis
H1763 Hampey
c.1
193226
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